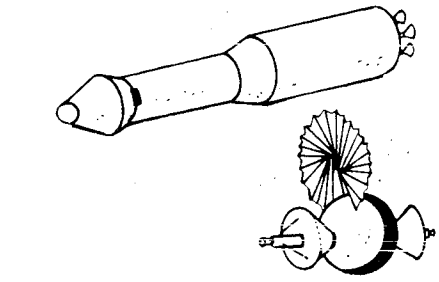
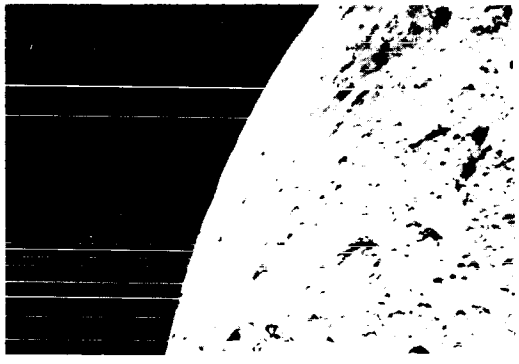
PHASE II PROGRESS REPORT

16 JANUARY 1962





# ORBITAL LAUNCH OPERATIONS

## VOL. V
## COMMUNICATIONS, ATTITUDE CONTROL AND SECONDARY POWER


VOUGHT ASTRONAUTICS

SPERRY-RAND CORPORATION

Control No. 64003
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(14 June 1961 - 16 January 1962)

Prepared for

George C. Marshall Space Flight Center
(Future Projects Office)
Huntsville, Alabama

Study of

# ORBITAL LAUNCH OPERATIONS

Volume V

## COMMUNICATIONS, ATTITUDE CONTROL, and SECONDARY POWER

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# INTRODUCTION

During Phase I of this study, Chance Vought made preliminary investigations in several electrical and electronic subsystem areas. These included:

Communications

Stabilization and Control

Auxiliary Power

At the conclusion of Phase I, it was realized that many problem areas existed and that it would be desirable to pursue the studies further toward the broad objectives of resolving those problems, optimizing systems relative to OLO requirements and outlining development plans needed to assure availability of operational hardware to meet the OLO development schedule.

In order to secure maximum penetration of those subjects in the limited time available, it was decided to invite members of industry who have special capability in those specific fields to participate with Chance Vought in the last phase of the study.

Sperry Rand Systems Group of Sperry Rand Corporation has done an over-all study of subsystems in orbital launch operations, including communications, attitude control and secondary power. The results of those studies are included herein as "Volume V, Part 1, Communications; Part 2, Attitude Control; and Part 3, Secondary Power."

In order to provide guidance for this work and to insure that their studies would be compatible with the work of Chance Vought, Sperry was furnished the following information:

1. Copies of the Chance Vought Phase I report.

2. Design information previously developed by Chance Vought relative to their study subjects.

3. Need to know to visit NASA and other agencies to secure background information.

4. A statement of work approved by the NASA project office stating the study objective, OLO program description and schedule, study ground rules, study areas, study approach and schedule. Emphasis was placed on those items of greatest interest to NASA and Chance Vought.

5. Liaison, including visits and meetings to review work progress and verify compatibility with the Chance Vought effort.

Information developed in these subcontracted studies has been incorporated into the Chance Vought work principally in the form of design characteristics of equipments and reliability estimates as they affect mission success probability.

# VOLUME V

## ORBITAL LAUNCH OPERATIONS

## PART I

## COMMUNICATIONS

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# SECTION I

# SUMMARY

## 1-1. INTRODUCTION

The Sperry Rand Corporation is supporting Marshall's Orbital Launch Operations Study (Vought Astronautics Contract NAS 8-853) by investigating the major Astrionic requirements for the program. The subject of this volume is Communications and Telemetry, which covers the work accomplished by the Sperry Rand Systems Group assisted by the Sperry Utah Company, Salt Lake City, Utah and the Sperry Development Center, Sunnyvale, California.

## 1-2. EVALUATION

The principal conclusions of this work are:

a. The communications and telemetry aspects of the OLO program are within the time scale for state-of-the-art development as conceived by Sperry Rand Corporation. No major technological break throughs are necessary to successfully establish Orbital Launch Operations.

b. Certain follow-on study work is absolutely necessary to firmly establish plans for successful Orbital Launch Operations. In the area of Communications and Telemetry these are as follows:

(1) The requirement for continuous communications must be established in depth and fully documented since present

indications are that this communication continuity is a major requirement for successful Orbital Launch Operations.

(2) The continuous system utilizes satellites to great advantage. Sperry Rand Corporation proposes to develop the specifications and schedules for the necessary satellites if required.

(3) Specifications and schedule for the complete communications system should be developed by June 1962 to meet 1965 circumlunar flight schedule.

(4) Realistic reliability studies are needed for the spaceborne communications arrangement. These should include breadboarding and testing of key elements.

Since the work is well within the capabilities of the SUCO and SDC Divisions of the Sperry Rand Corporation, it is requested that assignment of this work to Sperry Rand for the next funding period be considered. Budgetary cost information for this work is presented in Section 9.

c. Other recommended follow-on work is as follows. These items, while not placed in the absolutely necessary category, are still considered necessary from the standpoint of well balanced planning and proper initiation of work programs necessary to meet the objectives of the national OLO program on schedule.

(1) The NASA plans for integration of communications, telemetry, ranging and range rate determination through the use of the same equipment using different forms of modulation

should be incorporated in the OLO operation if timely. The work that JPL, MIT and STL have been conducting indicates a strong trend in this direction but considerable study is required to see whether or not any of the concepts will be generally implemented soon enough.

(2) After the broad communication plan has been partially detailed, a check should be made on uniformity and interchangeability in order to minimize the logistic problems aloft.

(3) Work should be continued concerning the use of low flying satellites as the transmission medium from outlying radar sites to the operation center.

(4) Investigate methods for reducing intentional interference from an enemy and also possible unintentional interference.

d. These programs are also recommended for Sperry Rand support of the Marshall Space Flight Center OLO Development Program. The funding requirements for each of these work items is presented in Section IX.

e. Some of the major technical conclusions which are of interest to the Marshall Space Flight Center as an aid in planning the work are listed.

(1) Continuous communications and telemetry is of paramount importance to the probability of OLO mission success and on the order of 75 KC of bandwidth can be provided by low flying satellites in the time period required.

(2) Switchable duplex circuits should be provided from the OLO operations center and computer to various portions of the OLO complex aloft in order to provide the necessary flexibility and allow for expansion if required.

(3) Low flying satellites may be used to supply data input paths to the operations center from outlying radar sites.

(4) The low flying satellites system is superior to the 22,300 mile high flying system by a launch weight ratio of about 21:1.

(5) RF equipment should be readily tunable to 3 or more frequencies in the same band.

# SECTION II

# INTRODUCTION

## 2-1. PURPOSE OF REPORT

The purpose of this report is to present the results of investigations conducted by the Sperry Rand Systems Group in the field of communications and telemetry. The Sperry Rand Corporation is conducting an across the board analysis of the various phases of Orbital Launch Astrionics to aid the Marshall Space Flight Center in developing a firm and well documented plan for establishing the highly advantageous technique of Orbital Launch. The work is being conducted in connection with Vought Contract NAS 8-853, and is being guided by Vought work statements and coordinating documents. Every advantage is being taken of the results obtained by Vought in the earlier phases of the contract.

The work included in this volume is an investigation of the various concepts and techniques for communications.

Figure 2-1 shows a model of the desired communications and telemetry system. High quality, reliable, and continuous communications is required between the:

a. OLF and Ground

b. OLF and Spacecraft

c. OLF and Astrotug

d. OLF and Extravehicular Man

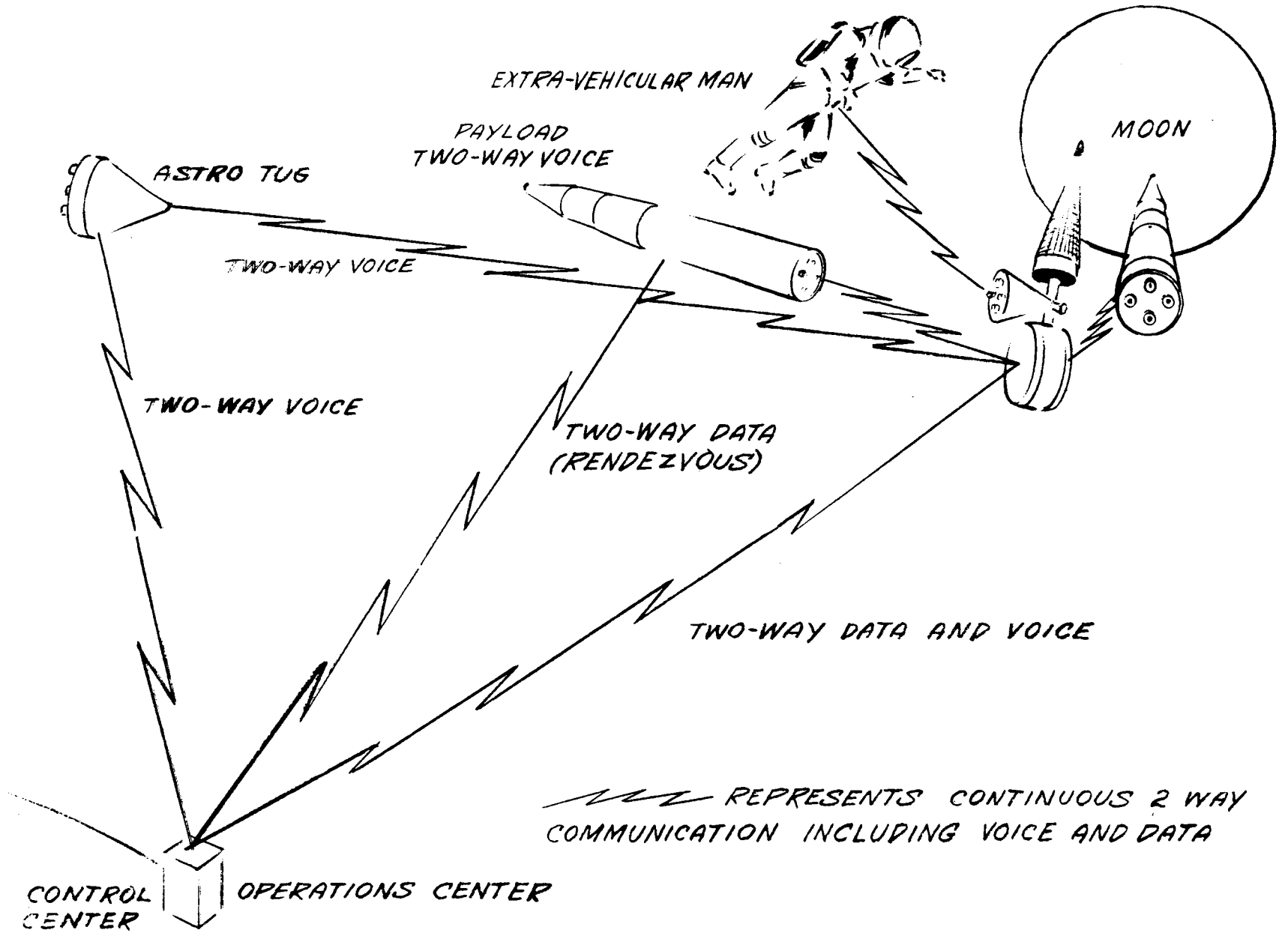



FIGURE 2-1 BASIC REQUIREMENT

This system must provide:

a. 2 way data links and voice channels

b. Data detection, processing and storage

c. Telemetry

Various concepts and techniques are investigated and a candidate system is derived by considering the following trade-off factors:

a. Performance

b. Overall cost

c. Reliability and life

d. Maintenance and Logistics

e. Weight, size, and power drain of spaceborne equipment

f. Schedule

The resulting candidate system that is described in this report has the following outstanding characteristics:

a. High quality, continuous duplex communications between all stations at all times.

b. Only one frequency band for entire system, thus providing:

(1) A minimum number of antennas

(2) Simple logistics

(3) Ease of maintenance

c. Every communications channel provides 50KC or more of bandwidth for multipurpose and flexible use.

This phase of the Astrionic work is being funded by Sperry Rand Corporation as a contribution to establish early the key engineering programs which are necessary to achieve success in this important endeavor. An outline of these key programs is presented in Section I of this volume. A more complete presentation may be found in Sections V, VI, VIII and IX.

# SECTION III

# BASIC CONSIDERATIONS

## 3-1. ORGANIZATIONAL PLAN

A discussion of the OLO communication problems is introduced by establishing several concepts by which one may test any conclusions as the work progresses. Following this discussion of concepts the differences and similarities between the OLO communication problems and the communication problems of other systems are outlined. This leads directly into the broad background of the candidate systems itself. The RF considerations will be presented first, followed by the telemetry terminal equipment. After presenting the best available facts and figures on the candidate system commensurate with the total time and effort available to date, the areas needing further study will be discussed.

## 3-2. BASIC CONCEPTS

### a. Line Switched Versus Message Switched Systems

Although in concept several investigators in the same environment may arrive at similar answers to the same problem, it is usually found that the grading of the answer is scaled to the experience of the investigator. For OLO communications it is important to take advantage of the major collective experience of the industry. For example, we may profit from established solutions to the problem areas encountered in our national and international

telephone systems, in aircraft control and other communication systems. These and other multiple user systems have something in common with the OLO operations. A requirement for "real time" operation is a fundamental of voice type operation. However, voice operation is sometimes reduced to non real time by handling recorded messages relayed through from one station to the next on a store and forward basis. On the other hand, there are cases where real time hot circuits are a necessity no matter how inefficient they may be. Fire alarm and police call boxes might be placed in this category. In other cases transmission capability can not be allowed to lie idle and must be shared with other users, for example, normal telegraph systems. The "real time" type circuit for multiple users will be referred to as a line switched system and the non real time circuit as a message switched or store and forward system. There are definite uses for each. However, there are occasions when the proponent of one type of system forgets of the existence of the other. An occasional test of the candidate system to see which system or combination of systems best satisfies the requirements is in order. The results of the subsequent study show the superiority of the line switched system for the OLO communications and as a subsidiary consideration duplex circuits are recommended.

b. Signalling and Supervision

As will be indicated later the OLO communication complex is a multiple user system. Multiple user systems require certain functions that may be forgotten if one has been working in two terminal, essentially single purpose systems. These added functions are signalling and supervision. For this discussion these terms are used in the broadest

sense. Signalling covers the area of addressing as well as the process of alerting the addressee of a call from another user. Supervision is related to signalling in that both may be considered in the category of necessary overhead in order to assure delivery or to inform the originator of progress in delivery. Supervision may be considered as information exchange between terminal points of a circuit that give the condition of the circuit and contribute only indirectly to the text or prime data being transmitted. Supervision includes such information as no-such-address, circuit busy, circuit out of commission, and message accounting numbering systems. These functions have quite a different form depending on whether one considers the line switched system, the message switched system or other systems that might be classed as hybrids or are not well differentiated from the point of view of how the routing is achieved. Very simple systems reduce the signalling and supervisory information to a priori information. As the system becomes more flexible much of this class of information varies from day to day, minute to minute, and must therefore be transmitted at approximately the same time as the text. There is a complete spectrum from bunched supervision once only (a priori) to fully distributed supervision with supervisory elements associated with each binary digit transmitted (error checks or correction). Another variable, on a somewhat orthogonal axis, is the way in which the supervision shares or does not share the path facilities of the text or prime data. A priori information is really one extreme on this axis, that is, certain exchange of information concerning the operation was made without using the circuit in question; the supervisory information did not share the path facilities of the text or prime data. Some portion of the supervisory

information is always handled in this manner. An example of a system that is toward the other end of this axis is a teletypewriter system on the poorest type of transmission line. Almost all the supervision is carried on through the use of spelled out words that are time shared with the text. On the other hand most telephone systems provided some measure of supervision simultaneous with the text although the great majority is time shared with the text. Many military communication centers provide separate circuits for the supervisory information and send address and text only on the main circuits. Attention has been given this concept since it is an area requiring detailed study before equipment specifications can be written.

c. Flexibility and Expandability

Another concept that should be used as a check on a communication system is its flexibility, including expandability. It has been shown that it is economical to plan for the future in communications and use a building block principle wherever possible. Can one satisfy one user today and another slightly different user tomorrow without completely rebuilding? In this regard one observes that there is little to distinguish between communications and telemetry, it is all data transfer. The capability used in communicating between machines needs little change to be able to exchange data between men. In the case of the OLO, there is a definite need for freedom in expansion. All of the requirements of 1967 cannot be forseen today.

d. Step Forward

A further concept that is a bit difficult to define is worth considering. That is, after investigating

utilization of existing concepts, equipment and installations and finding them inadequate one must think ahead somewhat beyond the solution of the immediate problem to be sure to take a step forward that is great enough to be worth the effort in breaking away from the more conventional scheme. In testing, one should perhaps go to a limit, even if absurd, and then retrench to a reasonable position.

## 3-3. DIFFERENCES BETWEEN OLO COMMUNICATIONS AND OTHER COMMUNICATION SYSTEMS

### a. Multi Station Complex, Continuous Communications

So much for basic philosophy, now consider what major differences and similarities there are between the OLO communications and other communication projects. Of course there are other space communication problems that are quite similar but most of those to this date do not include several "birds" of various natures in simultaneous and cooperative operation in orbit. Another difference between this and other space problems that can be rather severe in its consequences is that of continuous communications over an extended period of time (years). This follows the reasonable assumption that the OLF is to be permanent and not consumed by launch of the OLV. Both parts of this statement are differences from existing techniques. First, continuous communications implies a need for "real time" communications which has not been available previously. Second, the idea of having men aloft continuously for years places a permanence on the communication facilities that can alter the approach considerably over that of a one-shot approach. Reference to the figure 2-1, showing the overall communication complex, immediately demonstrates that there

may be several "birds" in operation simultaneously and provisions must be made for the concepts of signaling and supervision; for example, addresses are required that are not required in two-point communications.

b. Continuous Communications Vs. Continuous Tracking

Truly continuous communications is a severe requirement and requires a forthright attack that may not be a mere extension of the facilities used for other programs. For the present discussion, the requirement for continuous communications is divorced from that of continuous tracking of the OLF. In the event that continous tracking should be found to be a system requirement, the conclusions reached in this discussion would be somewhat modified. Reference will be made to the area requiring modification after the discussion excluding the possible modifications have been completed.

# SECTION IV

# CONTINUOUS COMMUNICATIONS, GROUND TO OLF

## 4-1. REQUIREMENTS FOR CONTINUOUS COMMUNICATIONS

Continuous communications requires at least one continuous channel and preferably several between the OLF and ground, the ground point being connected with the operations center with a set of channels of the same capacity as that from ground to OLF. Appendix A gives some of the details of this configuration as a simplified network. This provides background details for the difficult problem of continuous ground to OLF communications. Further, it requires at least one channel either to ground or to the OLF from any other "bird" in the near range of the OLF. For truly continuous operation this means that each of these circuits must be duplex so that one cannot lock another out by extended one-way use of reversible simplex. Appendix B gives an expansion of the terms simplex and duplex for those that may not be familiar with this terminology. It also explains the lock-out danger mentioned above. Simple switching is implied between the above circuits connecting the operations center to the OLF and interconnection at the OLF with the other "birds" in its immediate vicinity. Some of the network control problems which may be encountered in interconnection are indicated in Appendix C. This important concept of network control has been detailed in Appendix C since it is rather lengthy and may be well known to some of the readers. Since it will be demonstrated below that there is considerable expense involved in the OLF to ground link, a few of the highlights to justify

the need for continuous communications must be given. First, it may be argued falsely that no communications is required. Everything can be programmed as a sequence of operations clocked out of storage. This would be a true statement if everything were known to the desired degree of accuracy and reliability. It is definitely not true when human failures, equipment failures, weather changes, and other occurrences of unknown or only statistically known character are accounted for. Then one may say that in "rule by exception" one needs only to transmit the corrections to the stored program required by failures or insufficient knowledge and consequently only a very small portion of the total operation is consumed in sending corrections and discontinuous communications are satisfactory. Again this statement is probably valid so long as the variables in the program change smoothly, but may be false for step function changes to the program such as may be introduced by catastrophic failures of man or machine. The OLO is a very complicated operation involving the coordination of many, many functions and people and a failure to be able to transmit a "hold" just before launch of new crew members might require a $100,000,000 abort due to a failure in the OLF. One takes a calculated risk in assuming that any given time may be allowed for interruption of communications. In view of the length of the total operation and the total number of launches required, it becomes increasingly risky to leave holes in the communications. Another reason for communications continuity may be drawn from experiences at sea or in the air. It concerns the logging of information about the voyage. For any voyage in which the log book is salvageable from the wreckage, a self-contained log is satisfactory. However, remote logging is a necessity if one has any reason to believe that circumstances may obliterate the information in the self-contained log. If the instant of obliteration is not predictable,

continuous entries into the remote log are desirable. If it is believed that circumstances leading to the catastrophe might prevent similar catastrophes in the future, one has a rather reverent respect for the concept of an adequate log book. For this purpose, the word "log" has been used in a very broad sense and includes information entered in the records by the machine complex as well as by the men aboard.

After justifying continuous communications on other grounds, one finds that there are many bonus effects accruing after the basic objectives are satisfied. Operational convenience is one of the simplest to understand. Instead of restricting certain functions of conferences to an erratic come-and-go timetable, scheduling can be done without restriction.

## 4-2. EXTENSION OF EXISTING FACILITIES

In order to determine the magnitude of continuous communication type operation, the following brief analysis is given based on an extension of existing ground facilities. The chart of figure 4-1 gives the time-in-view of the OLF for ten selected locations on the earth. Fifteen consecutive orbits of the OLF are shown which very nearly close after 24 hours. On the first orbit the time-in-view for Cape Canaveral is 9.1 minutes, for Canary Island it is 1 minute, etc. Near the bottom of the column are given the times-out-of-view for over 8 minutes. There are other outages not shown less than 8 minutes. At the bottom of the page are the total times-in-view for each orbit. For the locations chosen, path ten is the worst, where the time-in-view is 8.3 minutes and the time-out-of-view is 88 minutes. Subsequent orbits 16 through 30 do not exactly repeat but there are enough samples given here to give a representative picture of what may occur in any 24-hour period. The locations were selected with some care

but even so are probably only representative, not optimum. Disregarding the distribution of in-view versus out-of-view, one finds that during 30.8 percent of the 24-hour period the OLF is in view of the ten ground stations. From this it may be seen that many more stations are needed to affect continuous coverage. To see how many stations are required, refer to figures 4-2, 4-3 and 4-4. Figure 4-2 shows that a 350 mile OLF can see the point 1420 miles away from the point directly under it, assuming 4/3 earth radius and minimum elevation 5 degrees above horizon. Figure 4-2 shows that the OLF at 28 degrees North and 350 altitude could see a satellite at the same longitude and altitude at the equator. This fact will be referred to later. Figure 4-3 shows how individual coverages may be overlapped to fill in the deep cusps. At 2400 statute miles between stations cusps fall to ±11 degrees of latitude from the line of centers. Figure 4-4 shows how two rows overlapped the same amount in both axes as figure 4-3 are inadequate for ±28 degrees latitude coverage and how three rows are more than adequate. As a matter of fact, three rows allow for some inaccuracies in location. This shows that 36 stations uniformly spaced will be the maximum number required. This number gives an opportunity to estimate the costs of such a ground complex.

## 4-3. COST OF GROUND EXTENSION

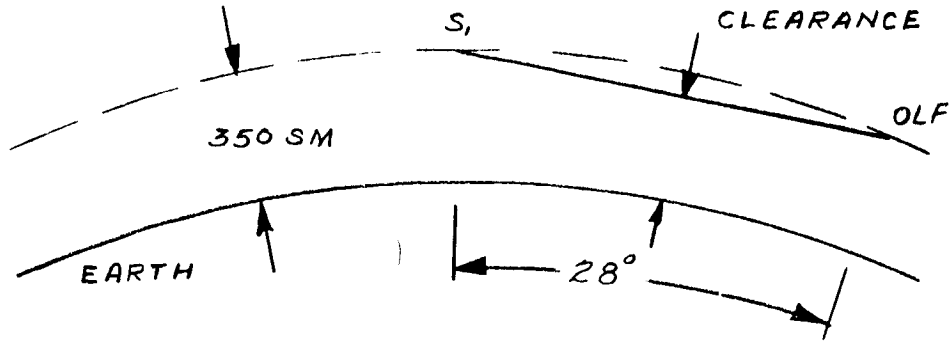
Of these 36 stations, a very rough estimate allows for 10 existing stations, 13 on land, and 13 at sea. Considering that whatever economies there are in outfitting existing stations will be absorbed in the stations at sea, one may see that to an order of magnitude the cost of ten years' operation of a ground net of 4kc bandwidth is $115,000,000. Figure 4-5 shows this estimate. Although this estimate needs refinement,

| Station / Orbit Period Number<br>λ = Long.(Deg.)<br>ø = Lat. (Deg.) | Continuous Time in View (Minutes) | | | | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 | 14 | 15 |
| Cape Canaveral, Florida<br>80,48W 28,48N | 9.1 | 9.1 | 7.3 | - | - | - | - | - | - | - | - | - | - | 7.8 | 9.1 |
| Canary Island (near)*<br>15, OW 28,48N | 1.0 | - | - | - | - | - | - | - | - | - | 7.0 | 9 | 9.1 | 8.9 | 7.8 |
| Ascension Island<br>14,41W 7,95S | - | 8.7 | 8.0 | - | - | - | 6.2 | 8.6 | 5.5 | - | - | - | - | - | - |
| Pretoria, South Africa<br>28,28E 25,75S | 6.6 | 9.0 | 9.1 | 9.1 | 8.9 | 5.8 | - | - | - | - | - | - | - | - | - |
| Mauritius Island (near)*<br>60E 20S | 8.6 | 7.7 | 8.2 | 9.2 | 7.0 | - | - | - | - | - | - | - | - | 6.0 | 9.0 |
| Woomera, Australia<br>136,79E 31,10S | 6.0 | - | - | - | - | - | - | - | - | - | - | 7.0 | 8.7 | 9.2 | 8.6 |
| Volcano Island (near)*<br>145E 27N | - | - | - | 4.5 | 8.6 | 9.0 | 9.0 | 8.7 | 6.0 | - | - | - | - | - | - |
| Hawaii<br>159,63W 22,16N | - | 8.2 | 9.0 | 8.5 | 8.5 | 9.0 | 6.8 | - | - | - | - | - | - | - | - |
| Society Island<br>150W 20S | - | - | - | - | - | - | - | 7.3 | 9.0 | 8.3 | 7.6 | 8.5 | 9.0 | 3.8 | - |
| Point Mugu, California<br>119,15W 34,12N | 6.3 | 8.3 | 8.3 | 7 | - | - | - | - | - | - | - | - | - | - | - |
| Continuous TOV values<br>(≥ 8 minutes) | 11<br>15<br>11<br>20 | 11<br>26 | 14<br>25 | 36<br>15 | 44<br>13 | 46<br>21 | 37<br>30 | 14<br>29<br>30 | 15<br>28<br>31 | 88 | 55<br>26 | 37<br>11<br>23 | 35<br>10<br>23 | 18<br>12<br>13<br>12 | 16<br>11<br>27<br>8 |
| TOTAL TIME IN VIEW | 37.6 | 51.0 | 49.9 | 38.3 | 33.0 | 23.8 | 22 | 24.6 | 20.5 | 8.3 | 14.6 | 24.5 | 26.8 | 35.7 | 34.5 |

*Ship Stations

Grand Total
TIV MIN
4451

Figure 4-1 Earth-Based Tracking Coverage of OLF for a 24 Hour Period

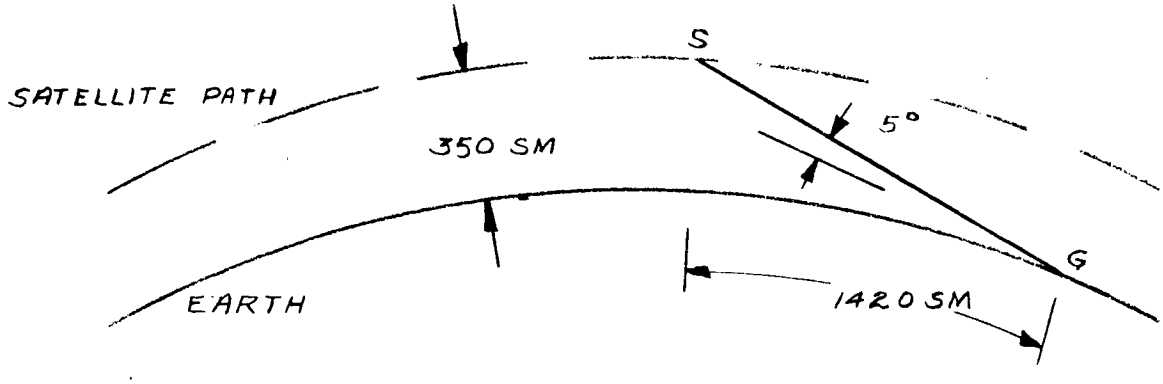



OLF TO EQUATORIAL SATELLITE PATH (OLF AT MAXIMUM LATITUDE)

S, - SATELLITE
OLF - ORBITAL LAUNCH FACILITY

A

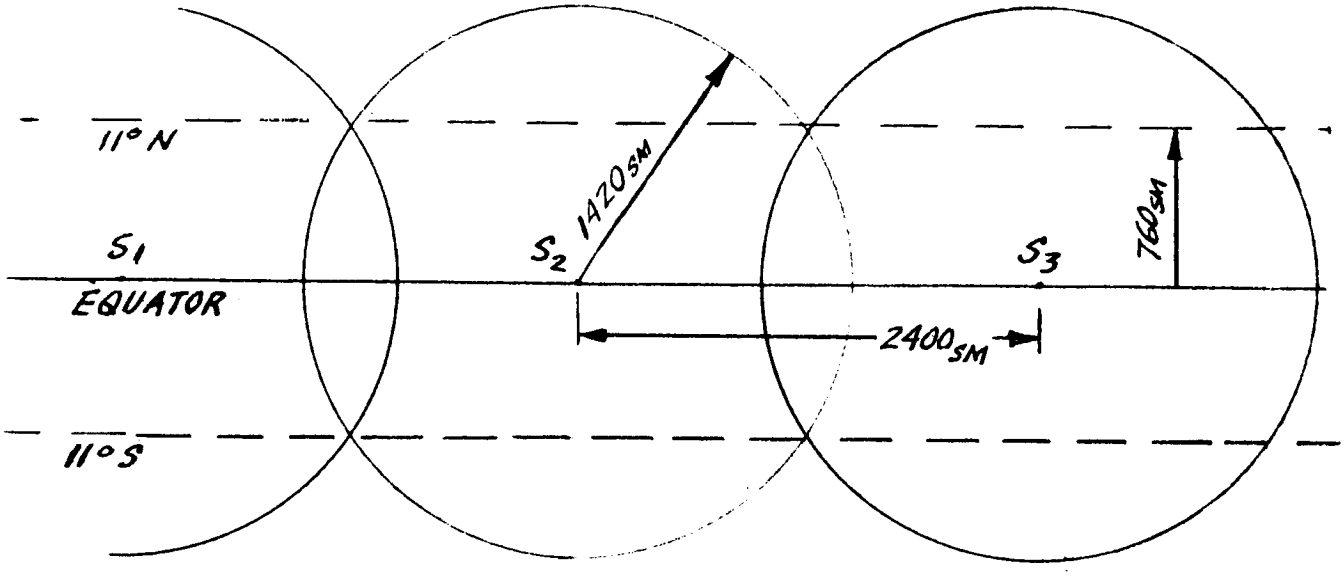
S - SATELLITE
G - GROUND




SATELLITE TO GROUND PATH
350 M ALTITUDE
5° ELEVATION ABOVE HORIZON

B

FIGURE 4-2 MAXIMUM LINE OF SIGHT PATH




$S_1$ SATELLITE 1
$S_2$ " 2
$S_3$ " 3

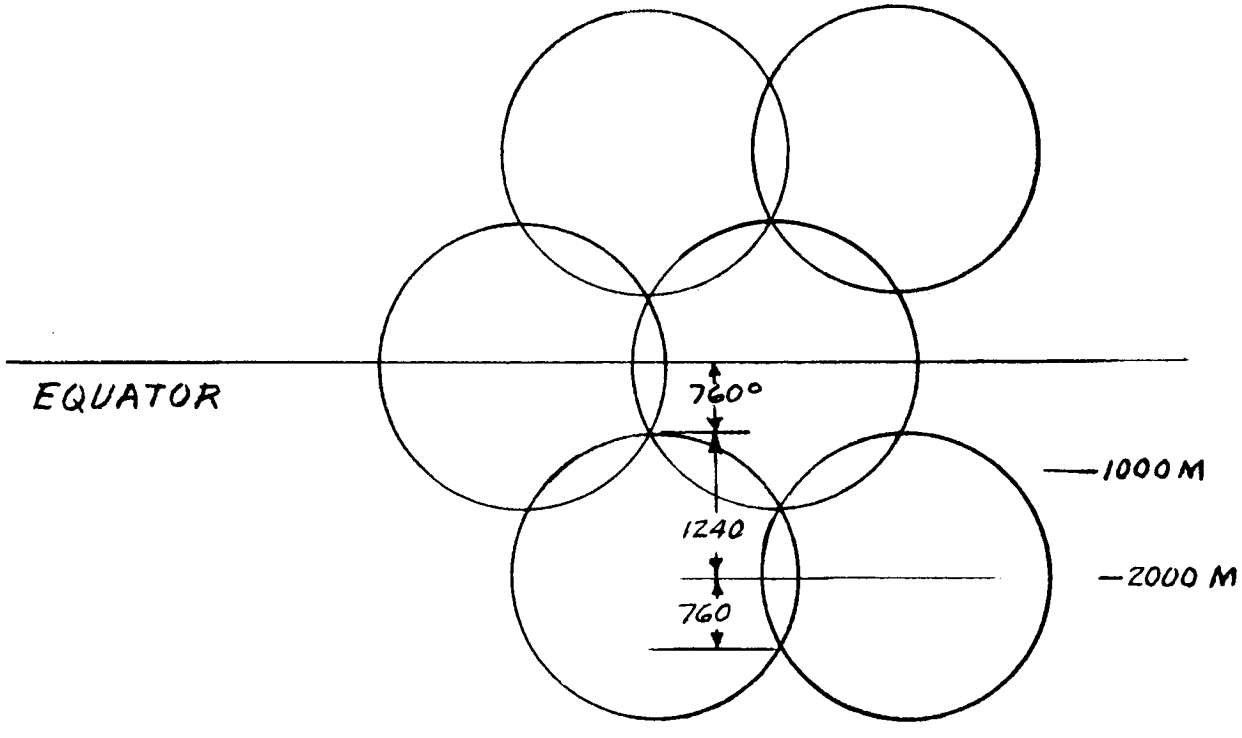
5° VIEWING ANGLE ABOVE TANGENCY

350 S. MILES ALTITUDE

2400 SM SPACING

FIGURE 4-3 EQUATORIAL SATELLITE RING GROUND COVERAGE

TOTAL 36 STATIONS
~ 10 EXIST
~ 13 LAND
~ 13 SEA




12 IN ONE ROW
2 ROWS FALL SHORT
3 ROWS MORE THAN ADEQUATE
(ALLOWS FOR CONSIDERABLE
INACCURACY IN LOCATION)

FIGURE 4-4 PATTERN FOR CONTINUOS COVERAGE

it shows that alternate schemes may be considered that cost on the order of hundreds of millions if they do a better overall job, such as more bandwidth, fewer propagation outages, or more operating convenience.

| | MILLIONS |
|---|---|
| 36 STA at $1,000,000 | $36 |
| LEASED LINES (10 YRS) 36 STA 5,000,000 MIN at $5 | $25 |
| 10 MEN/STA 10 YRS at $15,000/YR | $54 |
| | $115 MILLION |

Figure 4-5

Rough Estimate of Communications Ground Net Costs

## 4-4. USE OF SATELLITES

This leads one to consider the use of satellites. At first glance one assumes that shared use of communication satellites would be very economical. Further study shows that this may not be practical except as a means to connect together ground stations. However, a few limiting conditions may be understood from figure 4-6. The earth is well covered by the view from three high-flying satellites at 22,300 miles in stationary orbits. Without discussing the relative merits of the high-flying satellites one may turn momentarily to what other possibilities exist. A lower limit in altitude appears to be the height of the OLF orbit. As indicated in figure 4-3, such satellites would give a band 22 degrees wide around the earth. These must be mutually synchronous as must the 22,300 mile satellites. Two other possibilities immediately suggest themselves. Mutually synchronous orbits between 350 and 22,300 miles and random orbits, both polar and

| KIND | QTY | REMARKS |
|---|---|---|
| 22300M | 3 | ABSOLUTE MIN. QTY |
| 350 M | 12 | MIN. QTY FOR COMM WITH FENCE OF 3 GND STA |
| 350 TO 22300 M | 3<N<12 | |
| 7000 M | 40-50 | RANDOM |

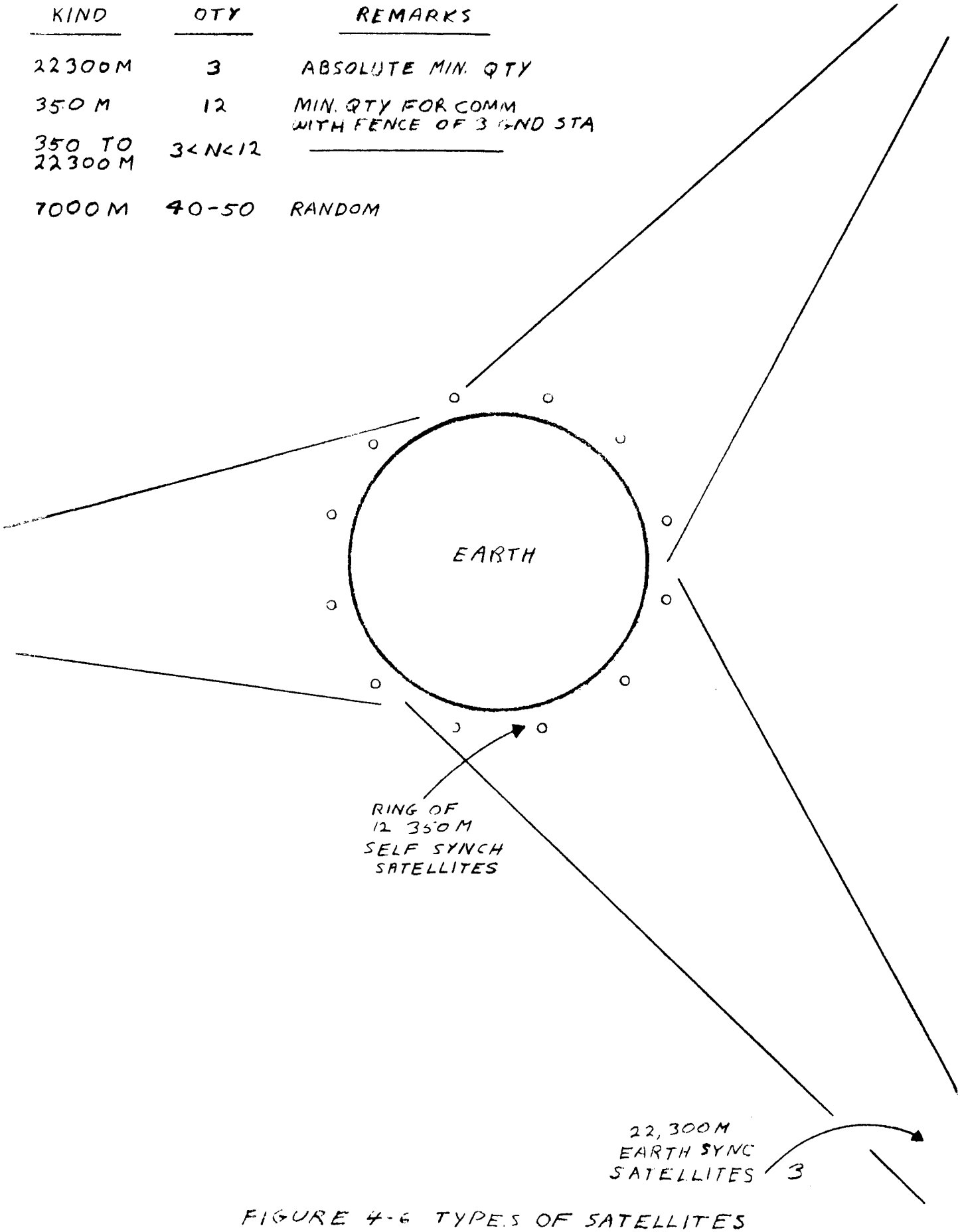



FIGURE 4-6 TYPES OF SATELLITES

equatorial at various altitudes. Figure 4-7 gives a few comparative characteristics of three types of satellites. Intermediate synchronous satellites have been omitted for the sake of brevity. The line discussing random satellites has not been fully completed since a check with Bell Labs indicates that OLF use of random satellites would probably require shutting down a pair of the ground stations that were originally designed to use the satellites. In addition, the acquisition and track column indicates that the random satellites would require considerable information about where to look next transmitted to and stored in a computer in the "bird" and upon loss of this information the system would have difficulty in learning how to heal the damage (loss of communications). The many coordination difficulties associated with random satellite systems indicate that the emphasis should be placed on comparing low-flying satellites versus high-flying satellites.

## 4-5. COMPARISON OF 350 MILE AND 22,300 MILE SATELLITE SYSTEMS

In order to make some comparisons of these two satellite systems, one must establish firm ground rules. The single rule that can change the picture more than any other has already been discussed but should be re-emphasized--continuity of communications.

It makes little difference in the following discussion whether one assumes shared facilities or new parallel facilities, certain masses must be injected into orbit to do this job. Sharing is particularly difficult if one considers that most satellite systems assume nonsymmetrical configurations with simple gear in the satellite and relatively sophisticated gear on the ground, weight and size

being more or less directly related to sophistication. If one wishes to utilize the satellite to communicate with another satellite, this nonsymmetrical system fails to bear dividends. Therefore, this analysis does not exactly parallel a ground-to-ground system. An accurate analysis should consider the cost of injecting the masses into the given orbits. A relative measure of this is given by the differences of the masses that must leave the ground. Assuming a $\Delta V$ of 7,000 ft/s to be required to go from a 350 mile orbit to 22,300 mile, it can be shown that with a solid fuel of $I_{sp} = 250$ the ratio of total mass to mass of payload plus structure is about 2.5.

Normalized weights will be computed with reference to the chart of figure 4-7. The least common denominator is the weight required to provide a given bandwidth at a given signal to noise ratio at the longest range in line 1. The maximum path length for line 1 is 2400 miles satellite to satellite (the 1600 miles slant range to ground does not govern). The weight of the equipment required to terminate one end of the 2400 mile path will be taken as unity. The 22,300 mile satellite has a maximum path length of its altitude plus one earth radius or 26,300 miles. Since the power required goes as the square of the distance, the power for line 3 is $\left(\frac{22,300}{2400}\right)^2$ or 121. Since the weight may be considered to go directly as the power (for this power range and continuous power), the ratio of weights per path is 121 to 1. For line 3 there are four installation aloft, three satellites and the OLF at mass 121 against 12 satellites plus OLF at unity mass or 13. Accounting for the difference in altitudes and utilizing the mass ratio of 2.5 on the three 22,300 mile satellites gives the ratio 13 to 1029. This is not the correct number as will be

| Type | Qty | Pwr | Unit Mass Per Path | Nr of Terminals | Mass X Nr. Terms | Acquisition & Track From OLF | Mass Ratio | Rel. Mass Y Mass Ratio Adjusted |
|---|---|---|---|---|---|---|---|---|
| Synchronous 350 MI | 12 | Low 1 | 1 | 12 + 1 | 13 | Simple 1 Axis ±15°Max | 1 | 13 |
| Random 7000 MI | ~40 | Med | >1 <121 | ~40 | Inter-Mediate | Complex 2 Axis ±90° | >1 <2.5 | Complex |
| Synchronous | 3 | 121 | 121 | 3 + 1 | 484 | 1 Axis ±90° | 2.5 | 1029 |

Figure 4-7

Comparison of Characteristics of Three Types of Satellites

shown later since each of the 350 mile satellites must be capable of looking in 4 directions. This yields (12 x 4) +1 or 49 versus 1029 or a ratio of 21. This is quite a large number and shows that further attention must be given the 350 mile satellites to determine what details of implementation are required.

## 4-6. NON-COPLANAR SATELLITE

The most attractive low-flying satellite is one at the same altitude as the OLF but in a non-coplanar equatorial orbit. A single satellite of this kind could aid the ground connection problem materially, since theoretically 12 ground stations at the equator would be in contact with it 100 percent of the time. Such a satellite could see the OLF at all times if maintained abreast of the OLF in an orbit of equal period, as shown in figure 4-2A. Such a concept may be extended to 12 satellites rather than 12 ground stations but this concept was rejected due to the difficulties in equatorial launching and maintenance.

## 4-7. COPLANAR SATELLITE

A derivative of the above system is a satellite in the path of the OLF. One leading and one lagging satellite spaced 30 degrees from the OLF, used as relays to each ground station, is an interesting concept. It triples the time in view over each station on the ground.

An obvious extension of this scheme is a limiting procedure. Extend these OLF relays to a ring of 12, one every 30 degrees. This reduces the number of ground stations required to a minimum of 3 placed in a north-south fence.

Appendix D develops practical hardware for the configuration as depicted in figure 4-8. The OLF is shown at the upper right in contact with the ring, numbered 1 to 12 and the ring in contact with the ground fence shown between 10 and 11. The ground station in contact with 1 is not part of the prime configuration, but gives direct contact to a given station, say the Cape, for certain intervals of time. A practical fence may include more than three stations. Figure 4-9 shows a fence using as many existing stations as possible, Antofagasta, Lima, Quito, Panama, Jamaica and the Cape.

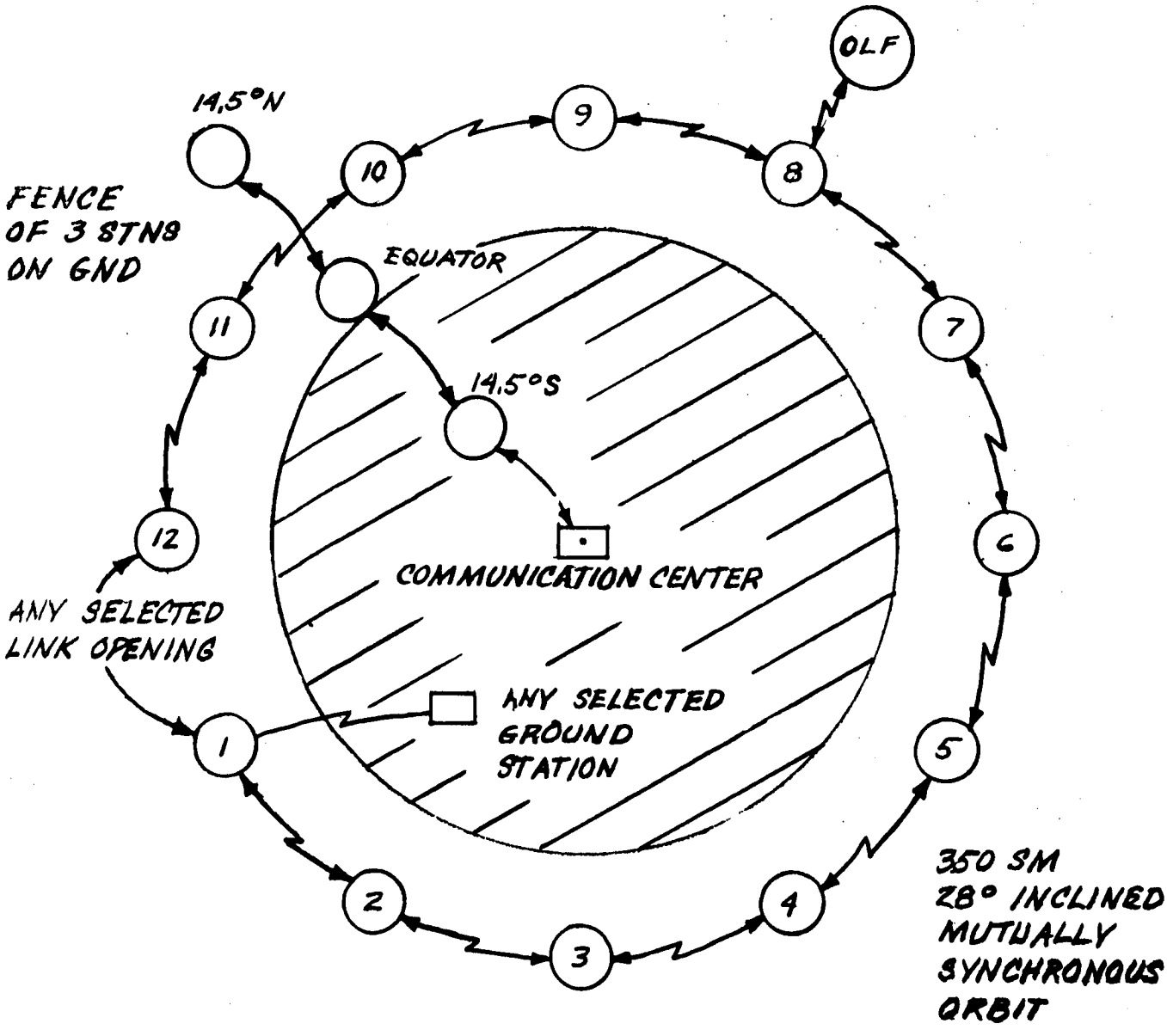



FIGURE 4-8 OLF- SATELLITE RING-
GROUND FENCE- CONCEPT

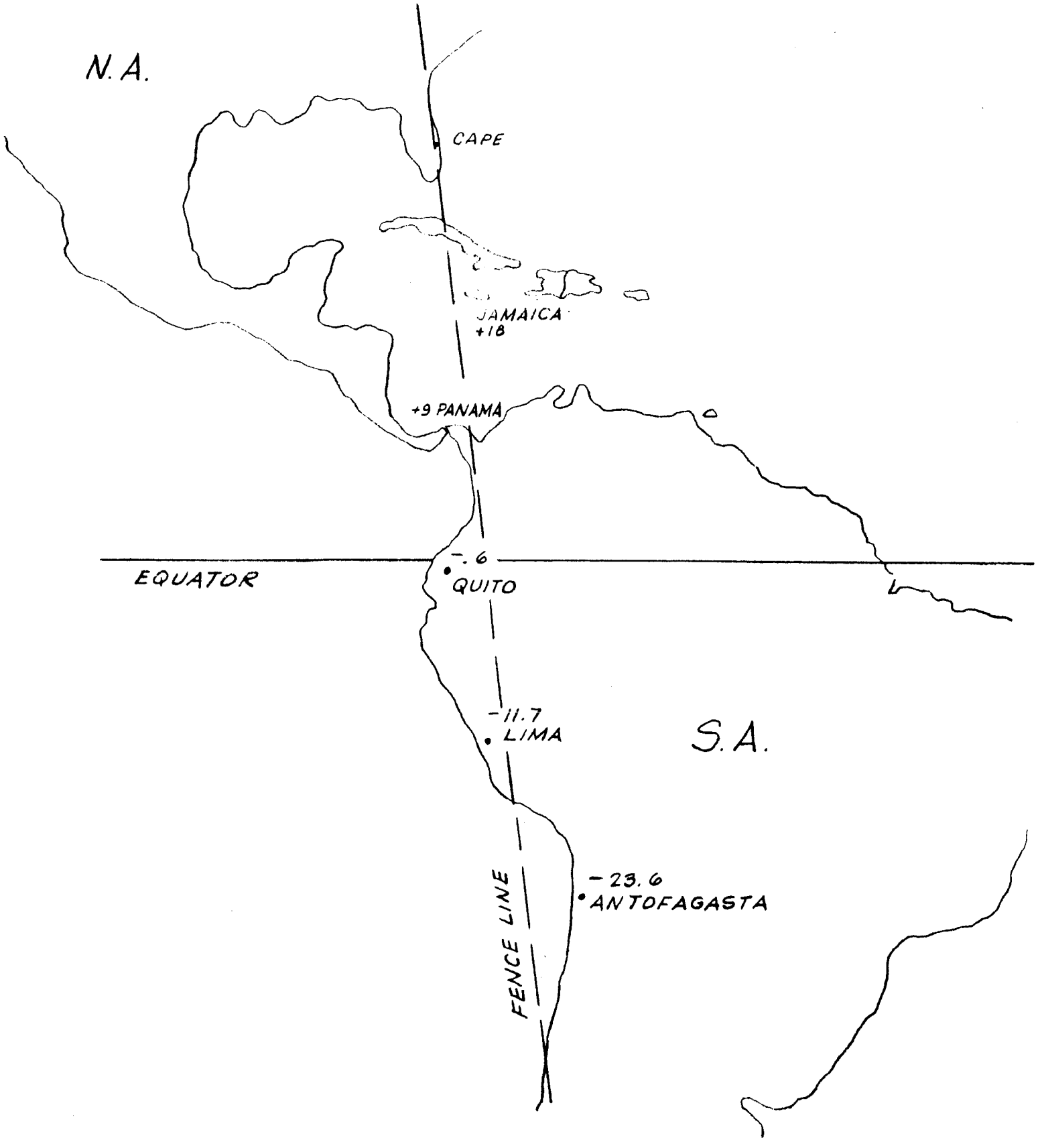



FIGURE 4-9 NORTH-SOUTH FENCE GEOMETRY

# SECTION V

# CANDIDATE OLF TO GROUND SYSTEM

## 5-1. SATELLITE RING

Having established a possible means of continuous coverage through satellite relays it is of interest to have a look at such a system in further detail. First, the chances of launching such a system on time with sufficient reliability to meet all the OLO requirements is proven by the basic ability of launch and rendezvous OLO components of all kinds. The total weight of 12 satellites is on the order of 6,000 lbs which could be boosted with other equipment to the OLF and subsequently each satellite may be given a small boost, of $\Delta V$ of about 50 ft/sec, at regular intervals in order to space the satellites ever 30 degrees around the earth. This would form a ring which slowly rotates with respect to the OLF. Directive antennas have been used on the satellites since the complication of orientation is considered well worth while in order to be able to achieve relatively wide bandwidth at a good signal to noise ratio. The components of the candidate system may be summarized with reference to figure 4-8.

## 5-2. GROUND FENCE

The ground fence is expanded to 6 stations each having UHF equipment looking up to the ring and HF equipment up the fence to the Cape as further described in Appendix D.

The block diagram of figure 5-1 shows the equipment required in each of the 12 satellites of the ring. There are four sets of duplex radios in each satellite, one looking forward in the ring, one looking backward in the ring, one looking up to the OLF, and one looking down to the ground fence. All the equipment is in the 2300 mc band. Eight foot dishes are used for the paths around the ring, while a two foot dish is adequate up to the OLF and a slot antenna is adequate looking toward the ground. The one watt transmitters for transmission around the ring are operative all the time. The 4 watt transmitter looking to the OLF and the 2 watt transmitter looking to the ground operate on a 10 to 20 percent duty cycle. The calculations leading to the configuration of figure 5-1 are detailed in the Appendix E. The OLF equipment for connection into the satellite ring is shown at the top of figure 5-2. This shows an eight watt transmitter and a parametric receiver operating with an eight foot dish. Calculation of this circuit is also given in Appendix E. The remaining three circuits of figure 5-2 are discussed in Section 6.

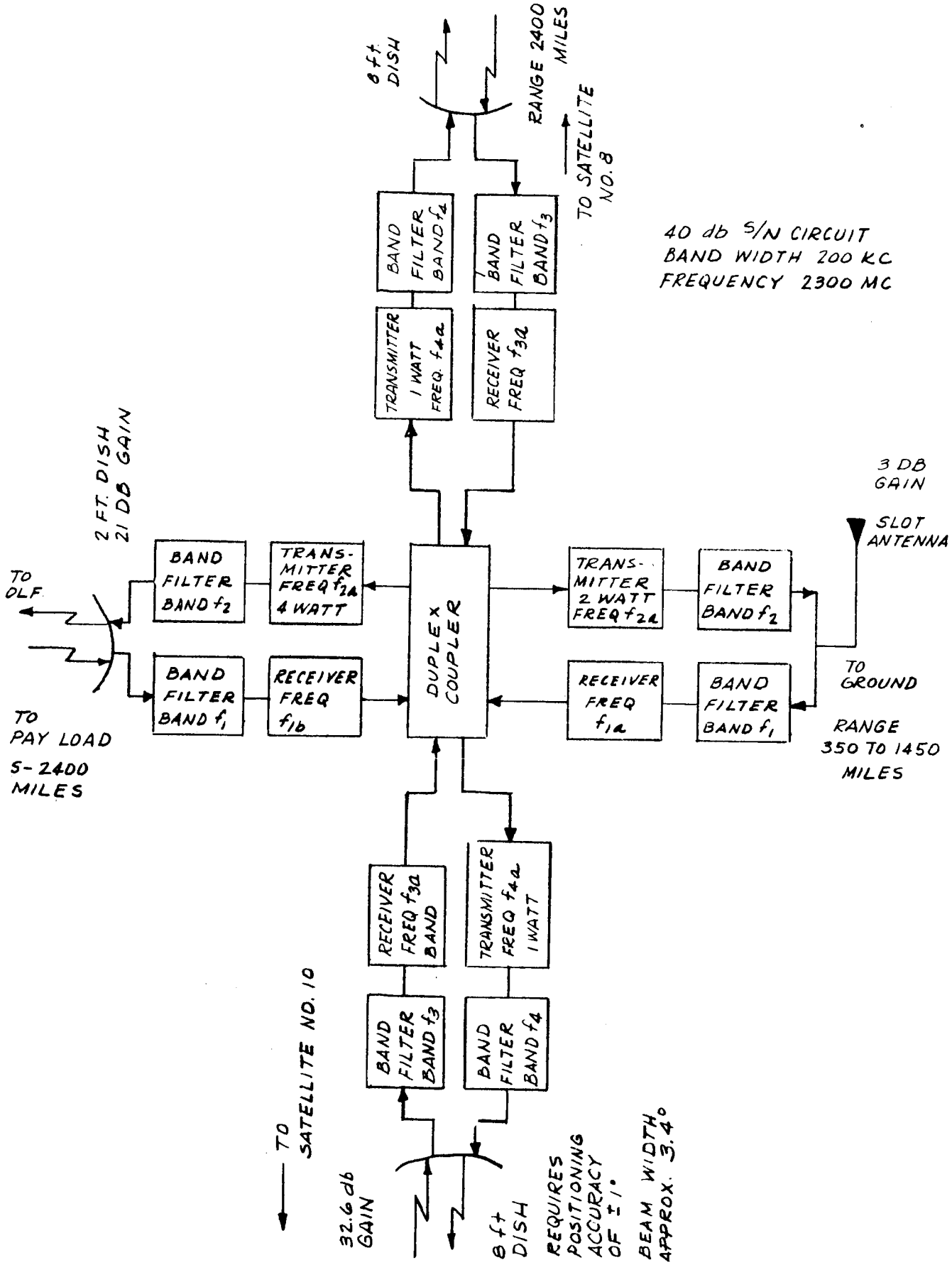



FIGURE 5-1 TYPICAL SPACE RELAY SATELLITE

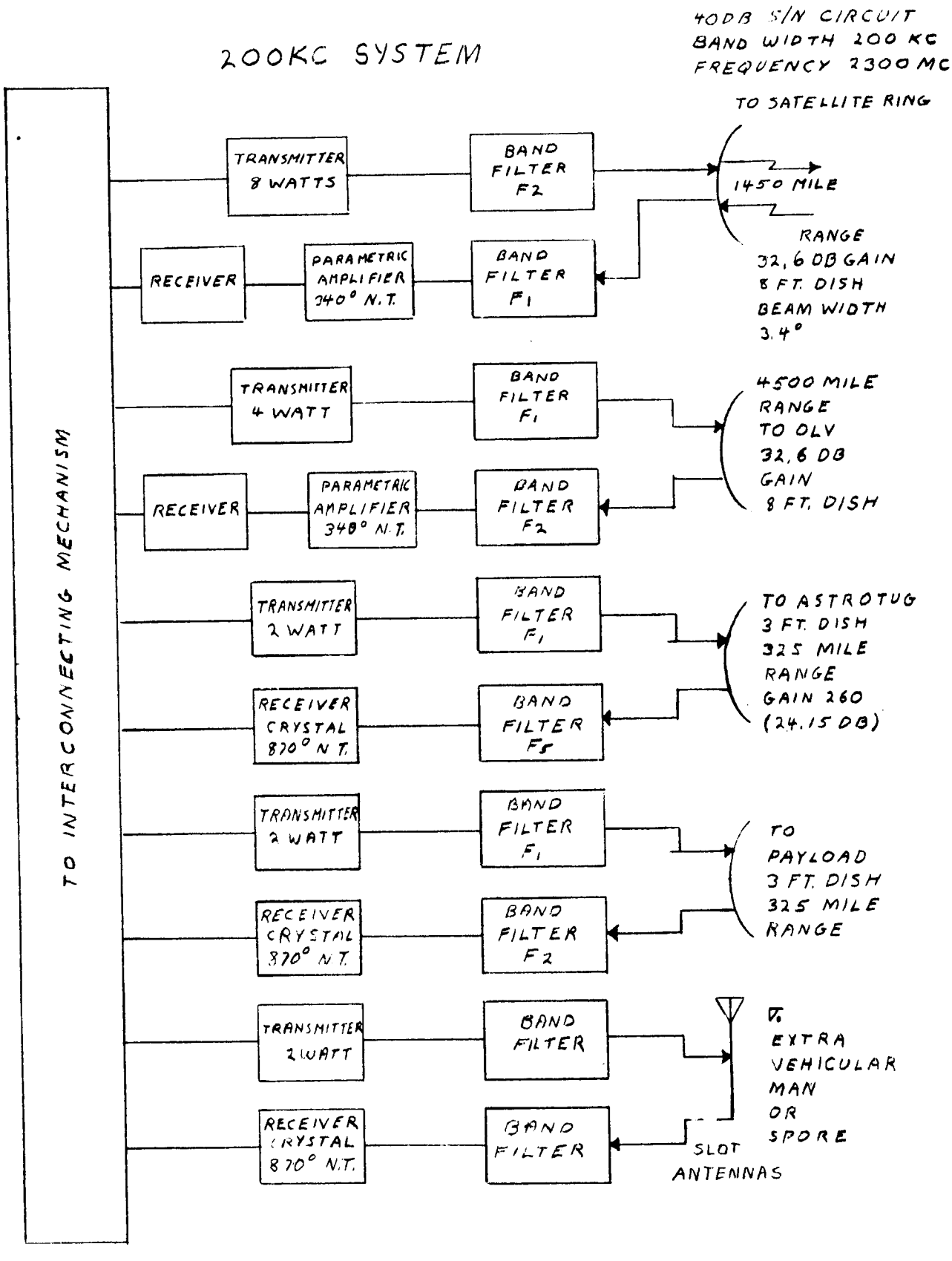



FIGURE 5-2 OLF EQUIPMENT

# SECTION VI

# OLF R-F EQUIPMENT

## 6-1. R-F EQUIPMENT

The OLF equipment has been summarized in figure 5-2. Supporting calculations are given in Appendix E. All of this equipment is in the 2300 mc band and requires 2 8-foot dishes and two three-foot dishes. Although every attempt should be made to keep the powers and receiver noise temperatures the same in order to increase the flexibility and built in redundancy (at reduced capability) one eight-watt transmitter, one four-watt transmitter and a pair of two-watt transmitters have been selected. Antenna and frequency switching should be provided, however, so that even the less sophisticated equipment could be placed on the most important circuit with somewhat reduce performance. In addition the modules of these equipments should be the same, for direct interchange in the case of failures.

## 6-2. CIRCUIT TO SPACECRAFT (S/C)

The circuit to the S/C is similar to that to the satellite ring except that not quite so much power is required for greater range since symmetry has been observed with the S/C, whereas in the case of the satellite ring a non-symmetrical circuit was utilized to keep the 12 satellites simpler and lighter. The range of 4,500 statute miles was selected to avoid the shadow area that might occur before the S/C would be high enough to be received by the

DSIF net. The geometry of this consideration is shown in figure 6-1. This figure shows the earth, the path of the OLF at 350 miles altitude, the path of the S/C up to 6000 miles altitude, and the near triangular cross-sections of shadow volumes that are below the horizon of the DSIF stations. Letters on the two paths, a to q, show the relative times so that relative look angles may be estimated. For simplicity the DSIF stations have been shown in the plane of the paths spaced 120 degrees apart. This approximation is close enough for the purposes of this discussion. The figure shows the DSIF shadow area covering the first 2000 miles of the path. This has no particular significance since the injection point might have any position relative to the DSIF stations. It may be seen that the DSIF shadow areas could be coincident with the first 4000 miles of the S/C's path. To eliminate the possibility of this problem the S/C may maintain communications with the OLF for the first 4500 miles of altitude and at this point transfer control to the DSIF net. This approach presents several advantages in that it reduces the required S/C communication payload and the required capabilities of the NASA ground complex while allowing a single station to maintain control of the S/C during initial powered flight.

## 6-3. CIRCUITS TO ASTROTUG AND PAYLOADS

The circuits to the Astrotug and payloads are identical and both are similar to the longer haul circuits. The range of 350 miles is a compromise between simplicity of equipment and early continuous communications for rendezvousing "birds" in the vicinity of the OLF. These might not be simultaneously required, particularly that

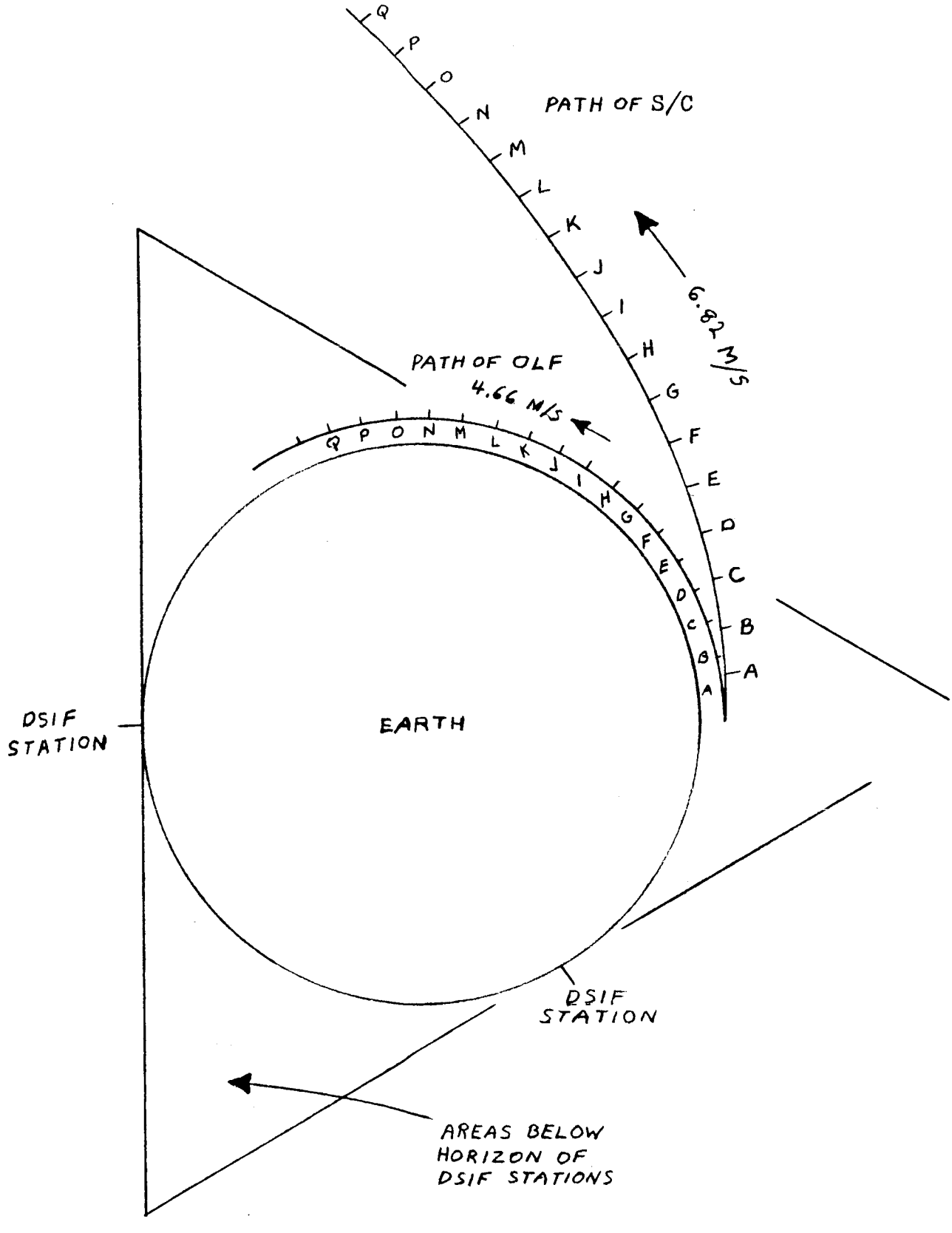



FIGURE 6-1 PATH OF S/C RELATIVE TO PATH OF OLF FOR ALTITUDES BELOW 6000 MILES

to the S/C, in which case the unused circuit should be considered as a spare. Again this points out that frequency agility and input/output switching are a firm requirement.

## 6-4. CIRCUITS TO THE EXTRA-VEHICULAR MAN

The communications to the extra-vehicular man may be handled primarily by telephone. It is assumed that there are safety lines to each of these men much like the man in the diving suit under water. However, for emergency, or in the case where he wishes to free himself of the hinderance of such a line radio should be used. This should not be in the low frequency bands, VLF through VHF, since very strong interfering signals may be received from the ground only 350 miles directly underneath. For this reason the UHF range has been selected once again. The fact that duplex operation may be obtained more easily at the higher frequency (since sharp filters are smaller in size) and compatibility with the rest of the system makes one more interconnection capability without added equipment if time sharing is allowed when other facets of the operation are shut down. A 10-milliwatt transmitter and a 30-db receiver is more than adequate for the job. The OLF equipment is the same as that for communications with the Astrotug or payload. A simple multiplexer can allow simple bio-medical data to be run on this circuit continuously if required, since the duplex capability is available. Duplex is also very important for the reason stated much earlier, that of prevention of unintentional monopolization of a reversible simplex circuit.

# SECTION VII

## MODIFICATIONS OF SATELLITE RING FOR CONTINUOUS TRACKING COMMUNICATIONS

As was mentioned earlier the study so far has separated the continuous communications problem from the continuous tracking problem. It is interesting to consider what use may be made of the satellite concept as proposed herein to send radar tracking data back to the central computer. Assume that a considerable number of radar stations widely dispersed throughout the world are made available to the project. These may be ships, or added land-based stations in order to get the necessary coverage. These stations individually have a limited use factor since, even if they are at the proper longitude, only less than one third of the OLF passes will be visible due to the changes in latitude of the OLF. Even though the satellite ring proposed is also only in view at any selected spot on the earth for the same percentage of the time, this portion of the time brackets the pass, or several passes of the OLF. This means that the satellite ring may provide a path which does not provide continuous communications to any given ground station back to the Cape, but is available as much as an hour or more before and after the given OLF pass. This is exactly when the data is generated that must be handled in real time or near real time. Some of the available bandwidth of the ring could be diverted for this job. The information would be available for both the operations center and the

OLF, if the OLF were to have sufficient computing capability to absorb the information. The total bandwidth can be divided with rather simple passband filters since there would never be more than one station, ground or OLF, simultaneously entering a single receiver in the ring.

# SECTION VIII

# OLF TELEMETRY DATA EQUIPMENT

## 8-1. OLF TELEMETRY DATA EQUIPMENT

The OLF Radio Frequency Communications and Telemetry circuits were presented in Section VI. Requirements were established for communications nets, OLF voice communications and data channels. In this section, the OLF telemetry channels are treated in detail: types of data to be handled are discussed and telemetry channel assignments estimated.

The OLF telemeter will be required to handle diverse data. Its capabilities will include data transmission and reception between both the ground complex and the S/C.

a. OLF Transmitted Data. The data to be transmitted by the OLF telemeter is as follows:

(1) OLF to Ground Complex

(a) OLF telemetry data showing the condition of the OLF and its crew and the results of scientific data acquisition. Specifically, this will include monitoring of OLF guidance, attitude controls, propulsion, temperatures, lift support parameters, bio-medical information, and miscellaneous equipment performance.

(b) Single frame video for the transmission of written displays, pictures of local conditions, and other images as required.

(c) S/C telemetry information, relayed to the ground complex via the OLF during countdown and the initial portion of powered flight by the S/C. Relay distances in excess of 4,500 miles do not appear feasible because of transmitter power requirements; consequently, DSIF must assume control early in flight. This approach presents several advantages in that it reduces the required S/C communication payload and the required capabilities of the ground complex while allowing a single station to maintain control of the S/C during initial powered flight. However, use of the OLF as a relay station for S/C intelligence will require that the data transfer capabilities of the OLF be expanded.

(2) OLF to S/C Transferred Data

(a) OLO mission data consisting of guidance and trajectory data for the S/C.

(b) Parameters required for checkout and countdown.

(c) Command signals relayed from the ground complex.

b. OLF Received Data. The data to be received by the OLF telemeter is as follows:

(1) Ground Complex to OLF

(a) OLO mission data consisting of guidance and trajectory data for the OLV during countdown.

(b) Commands selecting operating modes of the OLF telemetry equipment and performing other functions as required.

(c) Commands for relay to the S/C while the OLF is acting as a relay station.

(2) S/C to OLF Transferred Data. All S/C telemetry data for relay while OLF acts as S/C communications link.

c. Estimate of Telemetry Channel Assignments. Before conceptual hardware can be derived, some knowledge of telemetry measurement requirements is mandatory. Figure 8-1 presents a listing of representative measurements for a mission of this type together with estimated accuracy requirements and sampling rates. Figure 8-1 contains a total of 288 measurements listed under the broad categories of equipment monitor and evaluation, cabin environment, video, bio-medical, human abilities, and general scientific. About 95 percent of the measurements require only limited sampling rates. It is anticipated that OLO mission data will be presented to the telemetry system in serial binary form and that required rates will not exceed 1,000 bits per second.

## 8-2. OLF COMMUNICATIONS SYSTEM-OPERATING MODES AND EVOLUTION

The OLF communications system will possess a variety of operating modes. These modes include the telemetering of OLF performance and the condition of its crew, the acquiring of scientific data, the receiving of ground commands, the transferring of OLO mission data, and the relaying of spacecraft telemetry data.

In addition, requirements will exist for data storage and transmission on command. As the OLO program progresses, a gradual evolution can be expected in the functions monitored.

FIGURE 8-1

Estimate of Telemetry Measurement Requirements

A - Indicates 0.1 to 1%
B - Indicates 2 to 7%

OLF Equipment Monitor and Evaluation

| Function | Quantity | Accuracy | Maximum Sampling Rate (samples/sec) |
|---|---|---|---|
| Pitch acceleration | 1 | A | 20 |
| Linear acceleration | 3 | A | 20 |
| Angular rates | 3 | B | 20 |
| Attitude | 3 | B | 20 |
| Reaction jet control | 12 | A | 20 |
| Reaction jet valve position | 12 | A | 1 |
| Fuel flow (rate) | 12 | B | 20 |
| Fuel supply | 6 | B | 1 |
| Power supply voltages | 6 | AB | 20 |
| Power supply frequency | 4 | AB | 20 |
| Programmer signals | 10 | | 1 |
| Radio altimeter | 1 | A | 20 |
| Position data | 2 | A | 20 |
| External temperature | 8 | B | 1 |
| Inside air temperature | 4 | B | 1 |
| Equipment temperature | 20 | B | 1 |
| Meteorite impact | 1 | B | 20 |
| Command receiver signals | 10 | | 20 |
| Pitch jets | 1 | On-Off | 20 |
| Roll jets | 1 | On-Off | 20 |
| Yaw jets | 1 | On-Off | 20 |
| Miscellaneous | 38 | AB | 10 |

FIGURE 8-1 (Cont)

Estimate of Telemetry Measurement Requirements

| Function | Quantity | Accuracy | Maximum Sampling Rate (samples/sec) |
|---|---|---|---|
| General Warning (On-Off) | | | |
| Dangerous $CO_2$ level | 1 | | 1 |
| Cabin decompression | 1 | | 1 |
| Radiation warning | 1 | | 1 |
| Electrical system failure | 1 | | 1 |
| Environmental control system failure | 1 | | 1 |
| Attitude control failure | 1 | | 1 |
| | Total Functions = 167 | | |
| Cabin Environment | | | |
| Oxygen concentration | 1 | B | 1 |
| $CO_2$ concentration | 1 | B | 1 |
| Miscellaneous gasses | 6 | B | 1 |
| Total gas pressure | 1 | B | 1 |
| Gaseous flow | 2 | B | 10 |
| Cabin wall temperature | 2 | B | 1 |
| Cabin air temperature | 2 | B | 1 |
| Radiation rate | 4 | B | 5 |
| Humidity | 1 | B | 1 |
| $CO_2$ filter status | 2 | B | 1 |
| $O_2$ reserve | 1 | B | 1 |
| | Total Functions = 23 | | |

FIGURE 8-1 (Cont)

Estimate of Telemetry Measurement Requirments

| Function | Quantity | Accuracy | Maximum Sampling Rate (samples/sec) |
|---|---|---|---|
| Assumed Bio-Medical | | | |
| Electrocardiogram (EKG) | 1 | B | 320 |
| Phonocardiogram | 1 | B | 180 |
| Respiration | 1 | B | 20 |
| Electroencephalogram (EEG) | 3 | B | 120 |
| Galvanic skin resistance (GSR) | 1 | B | 20 |
| Blood pressure | 2 | B | 2 |
| Electro-oculograph (EOG) | 1 | B | 20 |
| Body temperature | 1 | B | 20 |
| Exhaled gases | 2 | B | 20 |
| | Total Functions = 14 | | |
| Human Capabilities | | | |
| Compensatory tracking | 1 | | 1 |
| Auditory vigilance | 1 | | 1 |
| Warning light monitoring | 1 | | 1 |
| Arithmetic computation | 1 | | 1 |
| Pattern discrimination | 1 | | 1 |
| Probability monitoring | 1 | | 1 |
| Scale position monitoring | 1 | | 1 |
| Miscellaneous | 10 | | 1 |
| | Total Functions = 17 | | |

FIGURE 8-1 (Cont)

Estimate of Telemetry Measurement Requirements

| Function | Quantity | Accuracy | Maximum Sampling Rate (samples/sec) |
|---|---|---|---|
| Assumed Scientific Date | | | |
| Scientific experiments will consist of cosmic ray sensors, scintillation counters, magnetometers, ion chambers, lyman alpha telescopes, electric field measurements, magnetic field measurements, plasma detectors, etc. No attempt is made to define this area in detail. For system definition, the following assumption is made: | | | |
| Miscellaneous | 67 | | 1 |
| | Total Functions = 67 | | |
| Narrow-band video - 100,000 samples per frame. | | | |

a. Evolution of Data Requirements During Phase II. As the program progresses, the emphasis on acquired data will change as follows:

(1) OLF During Ground Checks. During ground checks, all operating modes will be checked for proper operation.

(2) OLF Orbiting, Unmanned. Cabin environment and equipment performance will be closely monitored. Scientific data acquisition will not be commenced until the crew is on-board.

(3) Manned OLF. During the early period of manned operation, equipment performance, cabin environment, and bio-medical and psychological reactions will be heavily telemetered for detailed ground evaluation. Scientific experiments will be of secondary importance. This orientation will be required to perform field evaluation of all OLF equipment, detect environmental conditions dangerous to health, and detect dangerous medical conditions which have not been observed during previous programs but which might occur during extended orbital operations.

After the initial proving phase, psychological experiments will acquire higher priority since the over-all OLO concept requires that man be able to function effectively in the orbital environment for extended periods. During this same period, the feasibility of OLV checkout and countdown while in orbit will be proved.

b. Evolution of Data Requirements During Late Phase II. After the initial Phase II operations, equipment performance, bio-medical reaction monitoring, and cabin environment checks

will be performed on a scheduled basis. The acquisition of scientific data will now assume a role of importance secondary only to actual OLO operations, and more sophisticated psychological experiments will be performed.

c. Evolution of Data Requirements During Phase III. No marked changes in approach are envisioned during this phase. Two primary considerations will be made:

(1) Keep the system sufficiently flexible to allow the incorporation of a succession of scientific measurements.

(2) Retain sufficient capabilities to allow for modifications in the OLO program.

d. Evolution of Data Requirements During Phase IV and Subsequent Phases. It is projected that a redesigned communications package will be required at the beginning of Phase IV to keep the OLF facilities current with the evolving philosophy of OLO. A summary of telemetry requirements by operation is shown in figure 8-2.

e. Data Storage and Transmission. After the OLF has been established in orbit, continuous data transmission will be undesirable because of the extremely large quantities of redundant data which would be collected. Some data is ideally suited for storage with or without time correlation, while other data is of value primarily in real time. The requirements on the various types of data are discussed below.

FIGURE 8-2

Summary of Telemetry Requirements
By Operation

| | OLF Ground Checks | Injection to Orbit | OLF Unmanned in Orbit | Initial Manned OLF | Rendezvous and Docking | Orbital Launch Operations | Routine Operations (Phase II) | Routine Operations (Phase III) | Routine Operations (Phase IV) | Routine Operations (Phase V) |
|---|---|---|---|---|---|---|---|---|---|---|
| Equipment Performance and Monitoring | X | | X | X | X | X | X | X | X | X |
| Cabin Environment Monitoring | X | | X | X | X | X | X | X | X | X |
| Bio-Medical Monitoring | X | | | X | | | X | X | X | X |
| Psychological Monitoring | X | | | X | | | X | X | X | X |
| Receive Commands | X | X | X | X | X | X | X | X | X | X |
| Transfer OLO Mission Data | X | | | | | X | | | | |
| Relay of OLV Telemetry | X | | | | | X | | | | |
| Video Transmission (Slow Rate) | X | | | X | | X | X | X | X | X |
| Scientific Data Acquisition | X | | | | X | X | X | X | X | X |
| Operate Tracking Aids | X | X | X | X | X | X | X | X | X | X |
| Operate Voice Links | X | | | X | X | X | X | X | X | X |
| Two-Way Video (Real Time) | | | | | | | | | X | X |

(1) Bio-medical and Psychological Data. The crew members of a facility such as the OLF could not be expected to wear a bio-pack at all times. Therefore, a bio-unit will probably be provided which will acquire data on each man after it is put on. A psychological test unit of the same type will be employed. Then data could be collected at any free period throughout the day and stored for later transmission. No real-time correlation would be required.

(2) Scientific Data. Scientific data is primarily of a slowly changing nature and is gathered over long periods of time. To circumvent the continual ground processing of this data, it can be stored and transmitted later at a much higher rate to a single station. Real-time correlation will be required in this area.

(3) Video. Image storage will be required in this area to preserve desired pictures until ready for transmission.

(4) OLF Equipment Operation and Monitoring. After the OLF is established in orbit, ground checks of the OLF equipment will be conducted at predetermined intervals. No data storage will be required, since the primary application of this type of data is in real time.

(5) OLO Mission Data. OLO mission data will be received by the OLF from the ground complex and transmitted to the S/C. Storage will be provided by the OLF computer with transfer initiated by either local or remote command. The OLF telemetry system will not possess storage capabilities for this type of data.

Based on the preceding discussion, the concept has been developed that surveillance of the OLF will be performed by ground facilities at scheduled times. Real time OLF equipment operation or stored bio-medical, psychological, or scientific data will be transmitted by the OLF as selected by ground command. Measurement handling operation is reviewed in figure 8-3.

FIGURE 8-3

Measurement Handling Operation

| Measurement | Real Time | Store and Dump |
|---|---|---|
| Bio-medical | | X |
| Psychological | | X |
| Equipment performance and monitoring | X | |
| Scientific data | | X |
| OLO mission data | X | |

f. Evolution of S/C Telemetry Measurements. It is projected that the total number of telemetry measurements will remain constant throughout the OLO program. Equipment monitoring and evaluation will decline as confidence and experience increases, but will will be offset by an increase in scientific data acquisition. A graph showing the anticipated telemetry evolution is shown in figure 8-4.

g. Equipment Evolution and Hardware Schedule. It is projected that two generations of communications equipment will be required in support of OLO. First generation equipment

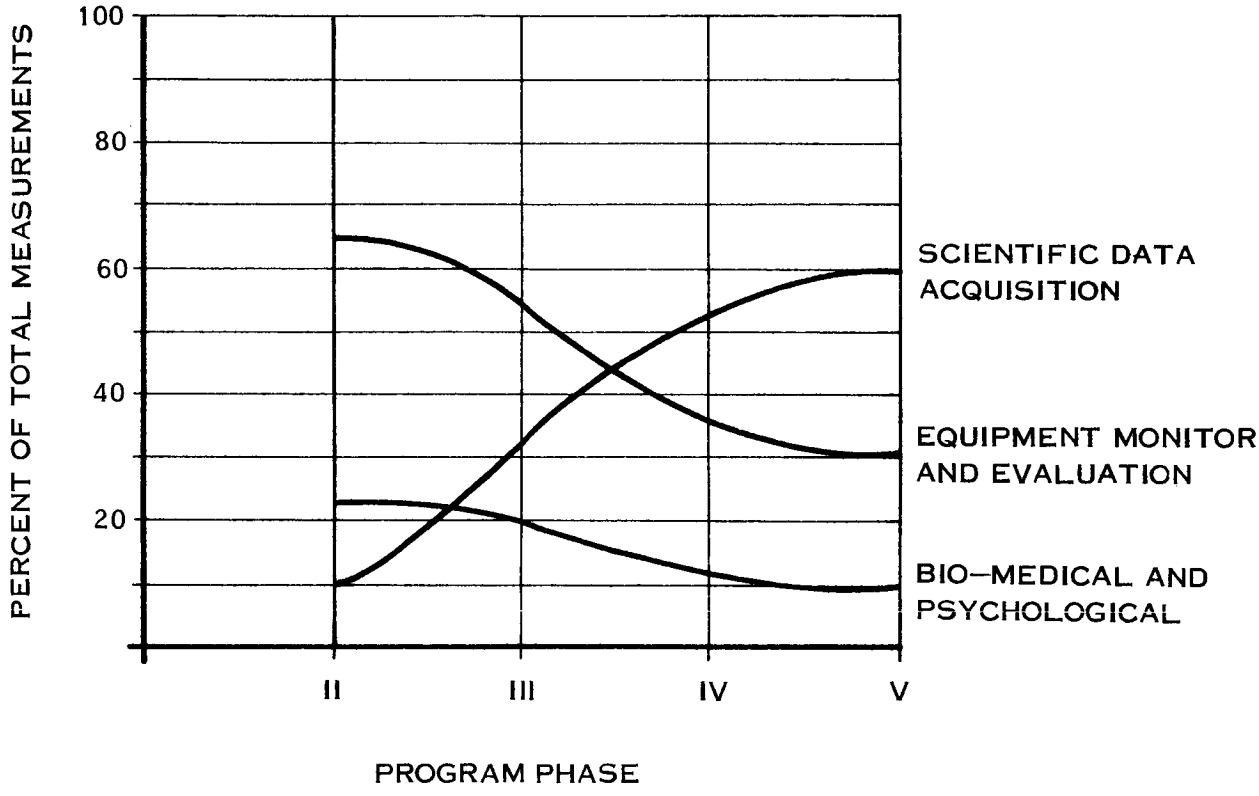
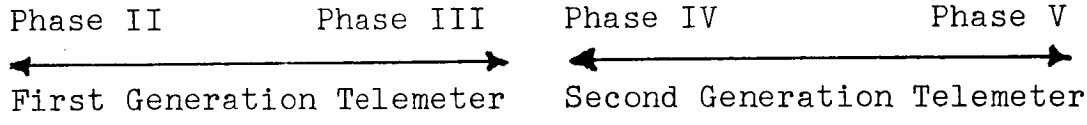



Figure 8-4 OLF Measurement Requirements

will be installed in the initial OLF. Changes in measurement requirements, refinements in orbital launch operations, a gradual shift of telemeter usage from an experimental to a research-tool, and OLF evolution will require a second generation communications system for the intermediate OLF. The second generation equipment will be sufficiently flexible to perform adequately throughout the remainder of the present program, including use with the ultimate OLF. Hardware evolution is as follows:




## 8-3. OLF RELIABILITY

The reliability of the OLF communications system will be an extremely critical factor. Because of the inherent high cost of supplying replacement units after the OLF is in orbit and because of the OLO operational requirements, a maximum effort must be made during all preflight operations, including design, fabrication, and checkout, to ensure that failure-free equipment is placed in orbit.

The reliability considerations for OLF communications equipment are presented in detail in Appendix G of this report. Component selection, burn-in, and extremely conservative circuit design will be employed to insure overall reliability. In addition, designs will incorporate multiple-use circuitry at the module, subassembly, and assembly levels wherever possible.

## 8-4. OLF MAINTAINABILITY

A detailed discussion of maintainability is presented in Appendix H. However, some additional specific comments and a summary of the Appendix H material is presented here for reading continuity.

Accessibility and fault isolation are two of the major problems to be considered. The design objective for the communications equipment will be fault isolation to the subassembly level through the use of self-check and ground-check techniques. For fault isolation at the module level, programming of the OLF checkout computer for subassembly testing and module isolation will be investigated. If the use of the checkout computer is feasible, module fault isolation by the computer will reduce the requirements for men as troubleshooting agents.

## 8-5. OLF SYSTEM FLEXIBILITY

Because of expected long-term operation and rapid evolution of system requirements as the OLO program progresses, considerable system flexibility must be maintained. Two basic approaches can be used to achieve this objective.

a. The telemeter can be provided with a memory unit and the program altered by local or ground command.

b. Man can act as a controlling agent and alter the program as required.

A completely automatic system for use with the OLF would be too complex and too costly. The optimum solution

appears to be a limited command program selection involving the basic telemeter only and man acting as agent for all measurement changes.

Sufficient system capacity will be provided to allow the addition of up to 30 percent of the original number of telemetry channels. The basic telemeter will provide the transmission capacity for this number of channels. The addition of a measurement will require only the insertion of a selected module or subassembly in the telemeter and wiring in the appropriate sensor.

The principle of on-board data reduction to limit transmitter bandwidth requirements and to reduce the amount of redundant data received on the ground was also considered. Extensive use of this approach was considered too complex and costly for the OLF communications system, especially in view of the previously developed concept of scheduled OLF surveillance instead of continuous data acquisition.

A single equipment design cannot be sufficiently flexible to be used throughout the OLO program. Program hardware requirements are discussed in paragraph 8-2.

## 8-6. ROLE OF MAN

The OLF communications system will not be required to operate in an unmanned condition except during the period immediately following initial injection of the OLF into orbit. During the remainder of the OLO program, the OLF crew will be expected to provide a fundamental portion of system programming, operation, maintenance, and repair. This program will obviate the need for extremely complex

equipment and will provide inherent system flexibility. However, man's responsibility in this operation is restricted by the limited time available for these duties.

a. OLF Telemetry System Programming. Figure 8-5 illustrates the concept for OLF telemetry programming. A fixed program under local or ground control determines the basic types of data being processed by the telemeter. Man will normally program only at the sub-level of basic data types (i.e., scientific data, bio-medical data, etc.) where measurements may be removed, added, or otherwise altered within the design limits of the system. This man-machine relationship appears to provide the best trade-off between complexity and flexibility for the OLF system.

b. OLF Communications System Maintenance. It is anticipated that most circuitry will be digital in nature, reducing significantly the problem of marginal operation. However, some analog circuitry will be employed, particularly in the signal conditioning equipment. This equipment will require periodic adjustment by the OLF crew. The design objective will be to provide measurement points such that a single unit of general-purpose test equipment common to other OLF equipment can be utilized.

c. OLF Communications System Repair. Fault isolation and repair were discussed in paragraph 8-4. Subassembly fault isolation will be performed by programmed self-checks and ground-checks, while module fault isolation will rely on the diagnostic ability of man in conjunction with checkout programming of the OLF computer. The candidate system is discussed in detail in Appendix F, where figure F-2 summarizes the system.

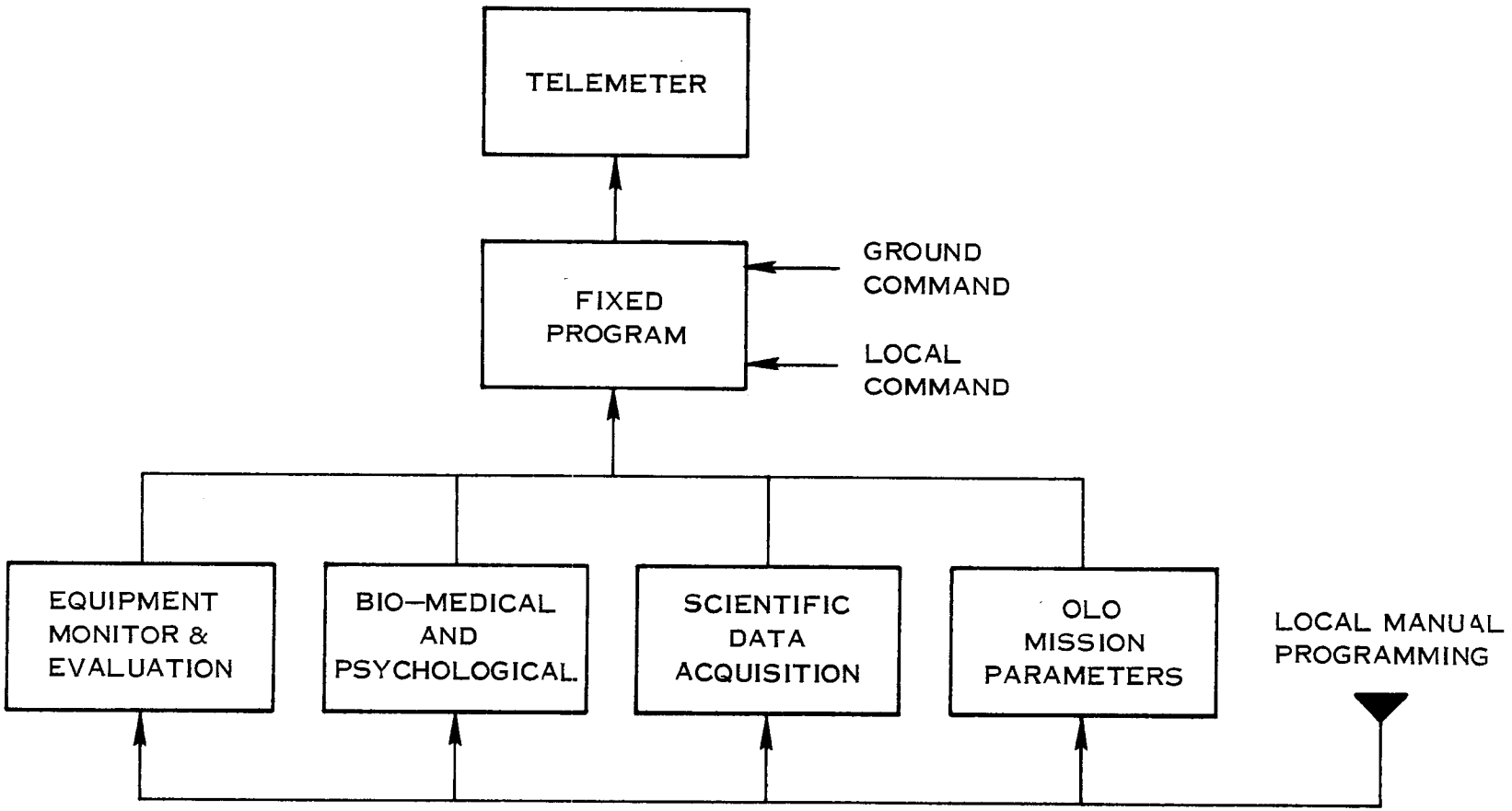



Figure 8-5 OLF Telemetry System Programming

# SECTION IX

# CONCLUSIONS

## 9-1. OTHER SYSTEMS

Before discussing the candidate system a comparison is made of several systems as shown in figure 9-1 in which the candidate system appears in the first column. A 36 ground station concept and an advent concept are shown in this comparison.

## 9-2. CANDIDATE SYSTEM

Four major aspects of the candidate system were discussed in Sections 5, 6 and 8 separately. These were the satellite ring and the fence which provided continuity of communication and telemetry from the ground operation center to the OLF. Two additional parts discussed, were the radio frequency equipment in the OLF and its immediate neighbors and the telemetry terminal equipment in the OLF. These four parts are now reviewed as a complete system. Figure 9-2 illustrates the candidate system including all of these four parts. Pictorially the telemetry terminal portion is not evident. The table of figure 9-3 has been provided summarizing the major equipments included in the fence satellites and the OLF. This table forms a basis from which the hardware cost estimates were made. Study, development, test, training and spare parts have not been

included in the table although they form elements of the total cost. These elements as well as the hardware have been called out in the program schedule of figure 9-4. This schedule basically covers the period from the beginning of the calendar year 1962 through the calendar year 1967. Although intermediate and ultimate OLF schedule is shown for reference purposes, none of the costs have been projected beyond 1967. It is important to note that the period of time from January 1962 to the end of the first quarter of 1965 is a very short time in which to provide thoroughly qualified hardware. The success of the program depends upon having prototypes in the latter part of 1963 in order to properly qualify equipment by the end of 1964. In order to achieve prototype units in the last half of 1963 extensive study programs must be accomplished during the first half of 1962. The number of hardware units have been indicated directly on the hardware schedule.

The costs are presented in graphic form in figure 9-5. The lower curve shows projected equipment and operating costs of the ground fence for the years 1962 through 1967 totaling 18.9 million dollars. The next higher curve shows the initial and equipment costs satellite ring totaling 26.3 million dollars. This figure does not include the boost cost in that consistant information was not available. The next higher curve shows the OLF communication and telemetry system equipment totaling 30 million dollars. The top curve is the sum of the lower three curves which results in a grand total of 75.2 million dollars for the period 1962-1967.

Figure 9-1

COMPARISON OF 3 BASIC CONCEPTS FOR CONTINUOUS COMMUNICATION

| Item | 36 Ground Station Concept | 12 Satellite Repeater Concept | Advent Concept (3 equally spaced satellites) |
|---|---|---|---|
| Relay Equipment - to provide continuous contact between OLF and A designated point on earth | A net of about 36 ground stations located accurately around the world. | 12 satellite repeaters located around the world at approx. OLF altitude. | Use Advent Facilities - both in space and on ground |
| Bandwidth Limit | About 3kc - for existing world network | 75kc proposed-could be several megacycles | Might be in megacycle range if channel is available |
| Cost | About $115 million including operating costs | About $75 million - not including booster costs for satellite repeaters | Total cost unknown. High Power Transmitter and Large Antenna needed on OLF for tracking Advent Satellites - provides difficult OLF stabilization problem |
| Availability | About 25% of required ground stations available. | 2 new fence stations plus additional equipment in 4 available stations. | Doubtful - if available on full time basis |

Figure 9-1 (Cont)

COMPARISON OF 3 BASIC CONCEPTS FOR CONTINUOUS COMMUNICATION

| Item | 36 Ground Station Concept | 12 Satellites Repeater Concept | Advent Concept (3 equally spaced satellites) |
|---|---|---|---|
| Other Considerations | Because of tracking requirements at each station it is doubtful that stations could be unmanned. | Would be available for other special communication requirements between earth and space vehicles located within the near vicinity of repeater. | Should be investigated more thoroughly. |

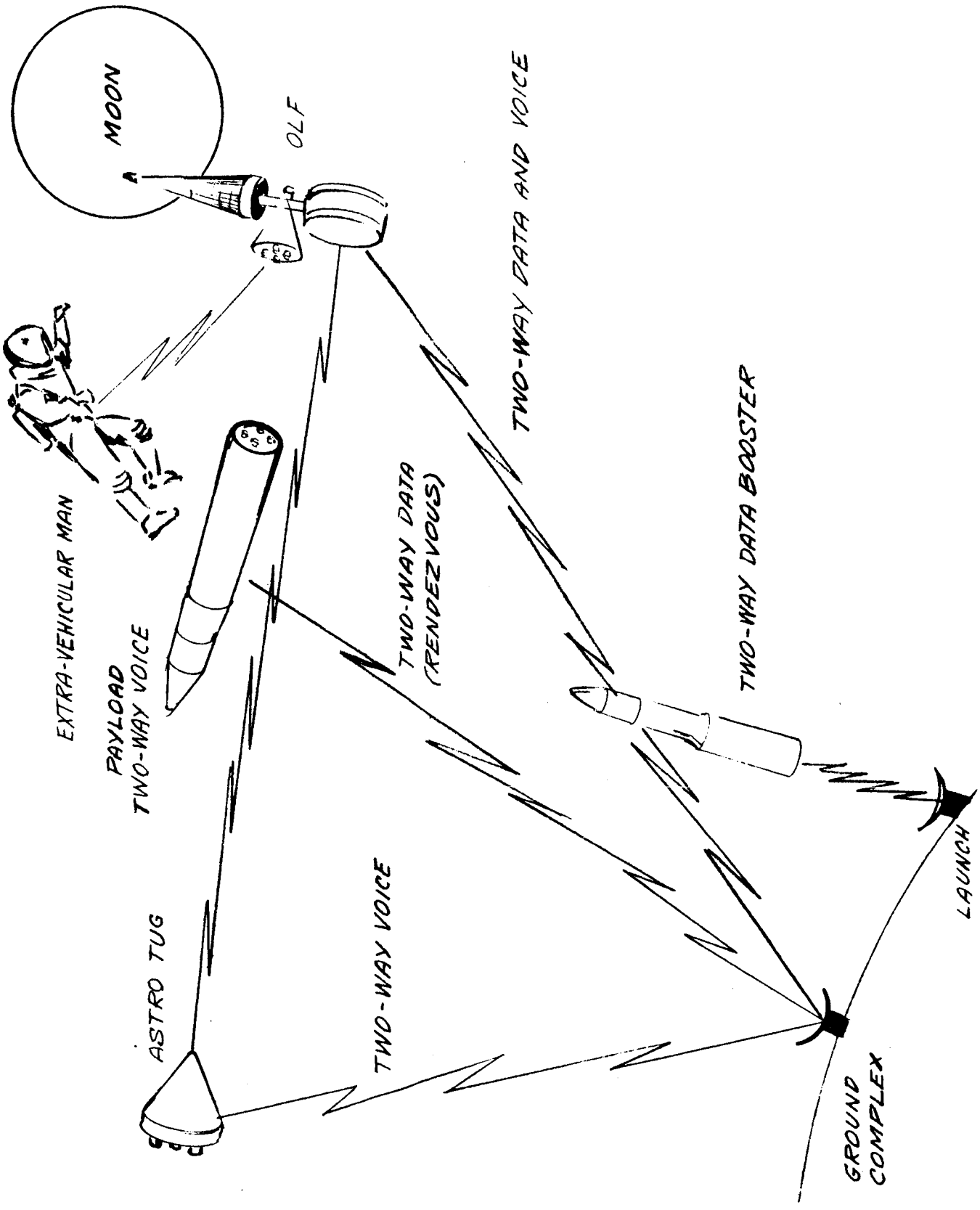



FIGURE 9-1A *OLO COMMUNICATIONS NET*

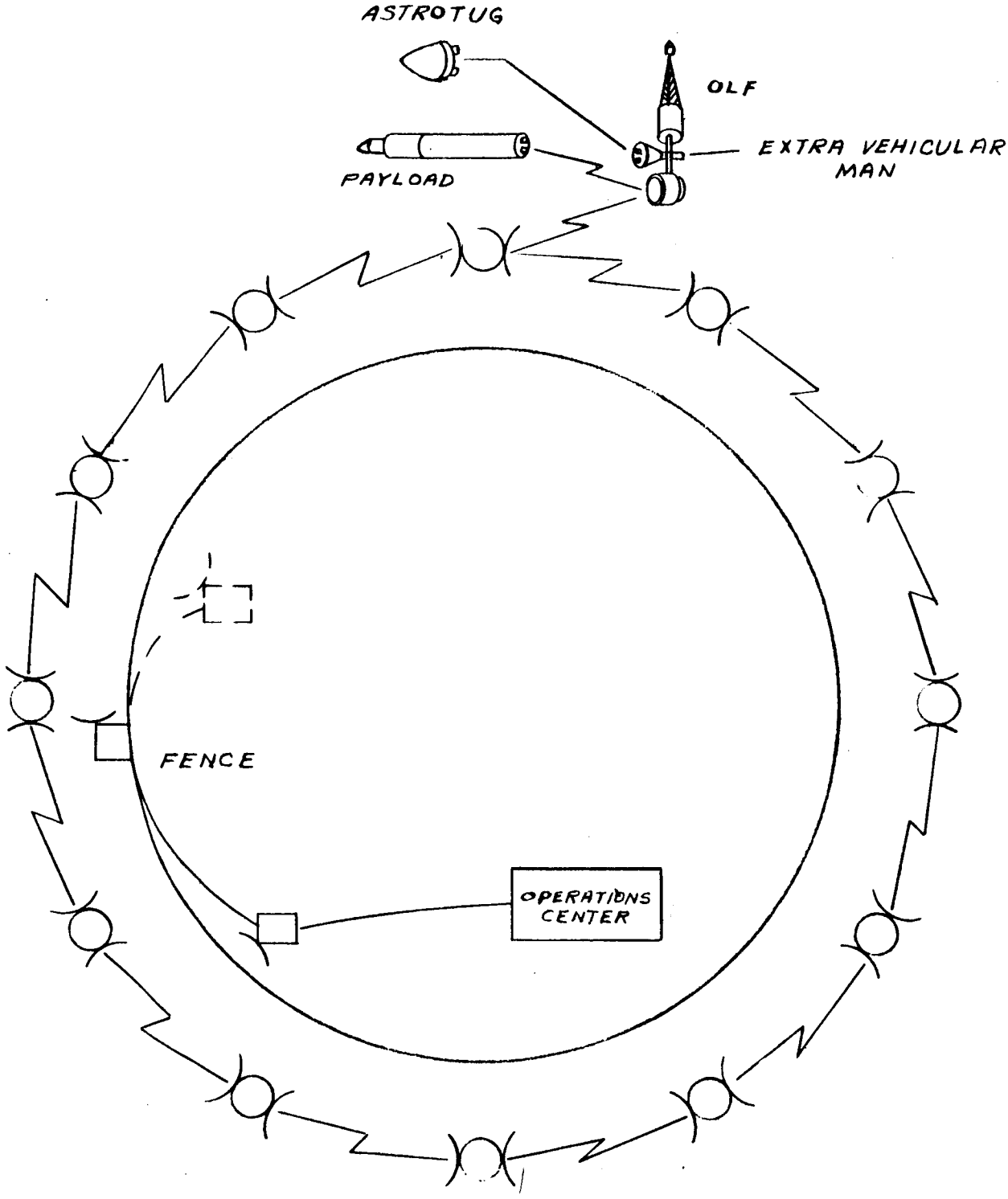



FIGURE 9-2 CANDIDATE SYSTEM

Figure 9-3

TABLE OF SYSTEM EQUIPMENT

| Station | Item | Qty/Stn | Num of Stns | Total Qty |
|---|---|---|---|---|
| Fence | 2300 mc Diplexer | 1 | 6 | 6 |
| | Tracking 48 DB Norm Ant | 1 | 6 | 6 |
| | 2300 MC Rec, NT 200°K | 1 | 6 | 6 |
| | 2300 MC Trans, 10 Watts | 1 | 6 | 6 |
| | Data Term. - UHF | 1 | 6 | 6 |
| | 2-30MC Log. Per. Antenna | 4 | 5* | 20 |
| | 2-30MC Trans. Filters | 6 | 5 | 30 |
| | 2-30MC Power Amps, 10KW | 6 | 5 | 30 |
| | 2-30MC Rec., NF 15DB | 6 | 5 | 30 |
| | 2-30MC Exciter | 6 | 5 | 30 |
| | Data Term., HF | 6 | 5 | 30 |
| | 2-30MC Preamp. | 2 | 5 | 10 |
| | Building & Power | 1 | 6 | 6 |
| Satellite | 8 Ft Dish Ant. | 2 | 12 | 24 |
| | Filter, 2300 MC, $F_3$ | 2 | 12 | 24 |
| | Filter, 2300 MC, $F_4$ | 2 | 12 | 24 |
| | 2 Ft Dish Ant. | 1 | 12 | 12 |
| | Filter, 2300 MC, $F_1$ | 2 | 12 | 12 |

*4 + 1/2 + 1/2 = 4 relays + 2 ends

Figure 9-3 (Cont)

TABLE OF SYSTEM EQUIPMENT

| Station | Item | Qty/Stn | Num of Stns | Total Qty |
|---|---|---|---|---|
| Satellite (Cont) | Filter, 2300 MC, $F_2$ | 2 | 12 | 12 |
| | Trans, 2300 Mc, 1 Watt | 2 | 12 | 24 |
| | Rec., 2300 Mc, Nt. 870°K | 4 | 12 | 48 |
| | Trans, 2300 Mc, 2 Watt | 1 | 12 | 12 |
| | Trans, 2300 Mc, 4 Watt | 1 | 12 | 12 |
| | Base Band Coupler | 1 | 12 | 12 |
| | Control Eqpt. | 1 | 12 | 12 |
| | Frame | 1 | 12 | 12 |
| | Power Supply Solar & Battery | 1 | 12 | 12 |
| OLF | 8 Ft Dish | 2 | 1 | 2 |
| | 3 Ft Dish | 2 | 1 | 2 |
| | Trans. 2300 Mc 8 Watts | 1 | 1 | 1 |
| | Trans. 2300 Mc 4 Watts | 1 | 1 | 1 |
| | Trans. 2300 Mc 2 Watts | 3 | 1 | 3 |
| | Baseband Inter-connections | 1 | 1 | 1 |
| | Internal Communications | 1 | 1 | 1 |

Figure 9-3 (Cont)

TABLE OF SYSTEM EQUIPMENT

| Station | Item | Qty/Stn | Num of Stns | Total Qty |
|---|---|---|---|---|
| OLF (Cont) | Filter, 2300 Mc | 8 | 1 | 8 |
| | Rec, 2300 Mc, Blt. 870°K | 5 | 1 | 5 |
| | Preamp., 2300 Mc, N.T. 340°K | 2 | 1 | 2 |
| | Telephone to Extra-Vehicular Man | 3 | 1 | 3 |
| | Telemetry Terminal | 1 | 1 | 1 |
| | Slot Antenna | 6 | 1 | 6 |
| Extra Vehicular Man | Telephone | 1 | 3 | 3 |
| | Rec. 2300 Mc NF 30DB | 1 | 3 | 3 |
| | Trans. 2300 Mc 10 Mw | 1 | 3 | 3 |
| Payload or Astrotug (Not Priced under OLF Package) | 3Ft Dish | 1 | X | 1X |
| | 8 and Filter | 2 | X | 2X |
| | Trans, 2300 Mc, 2W | 1 | X | 1X |
| | Rec, 2300 Mc, W.T. 870°K | 1 | X | 1X |
| | Telemetry Terminal, simplified | 1 | X | 1X |

Figure 9-3 (Cont)

TABLE OF SYSTEM EQUIPMENT

| Station | Item | Qty/Stn | Num of Stns | Total Qty |
|---|---|---|---|---|
| S/C (Not Priced under OLF Package) | 8Ft Dish | 1 | 1 | 1 |
| | Band Filter | 2 | 1 | 1 |
| | Trans, 2300 Mc, 2W | 1 | 1 | 1 |
| | Rec, 2300 Mc, NT. 870°K | 1 | 1 | 1 |
| | Telemetry Terminal | 1 | 1 | 1 |

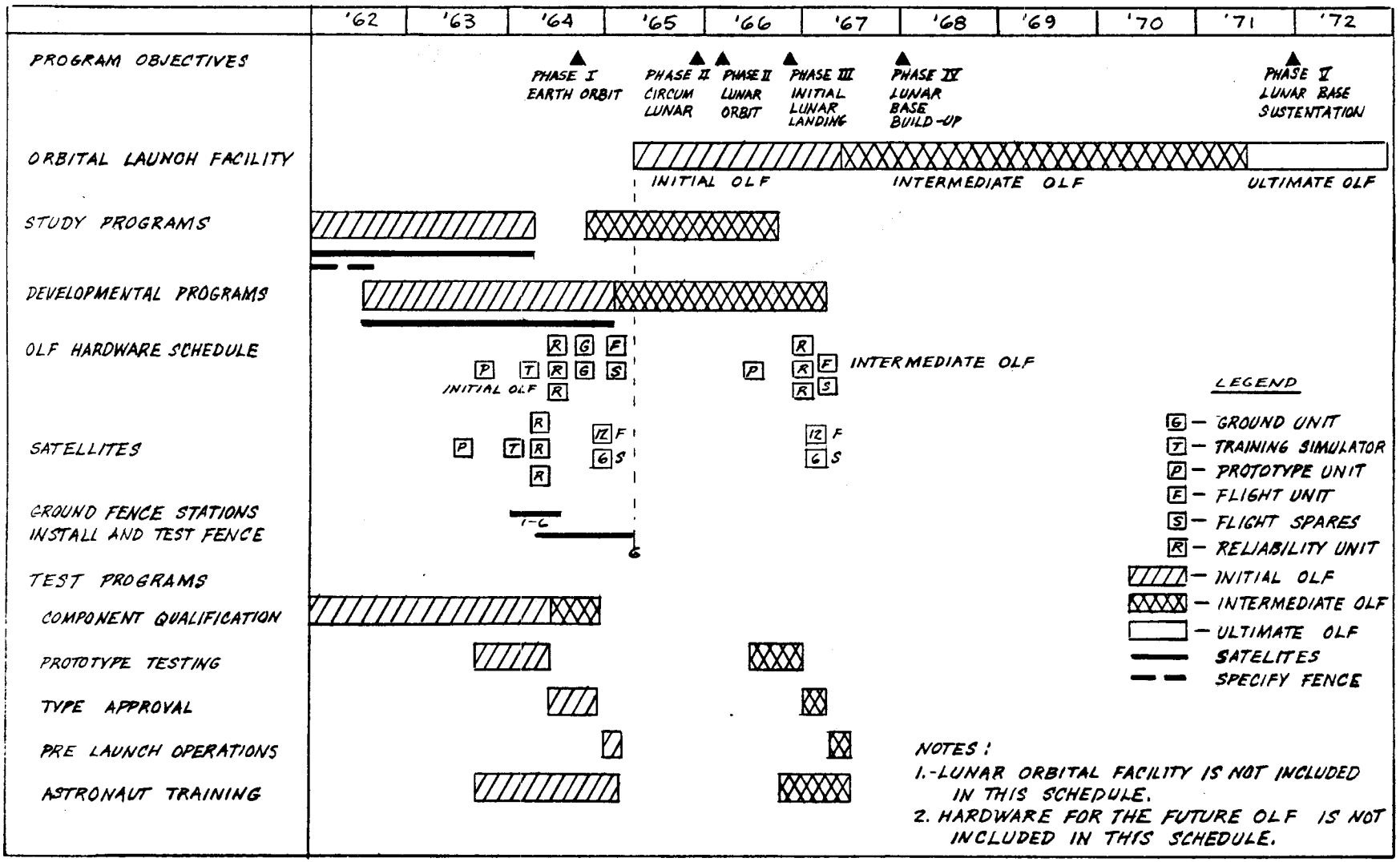



FIGURE 9-4 PROGRAM SCHEDULE TELEMETRY AND COMMUNICATIONS

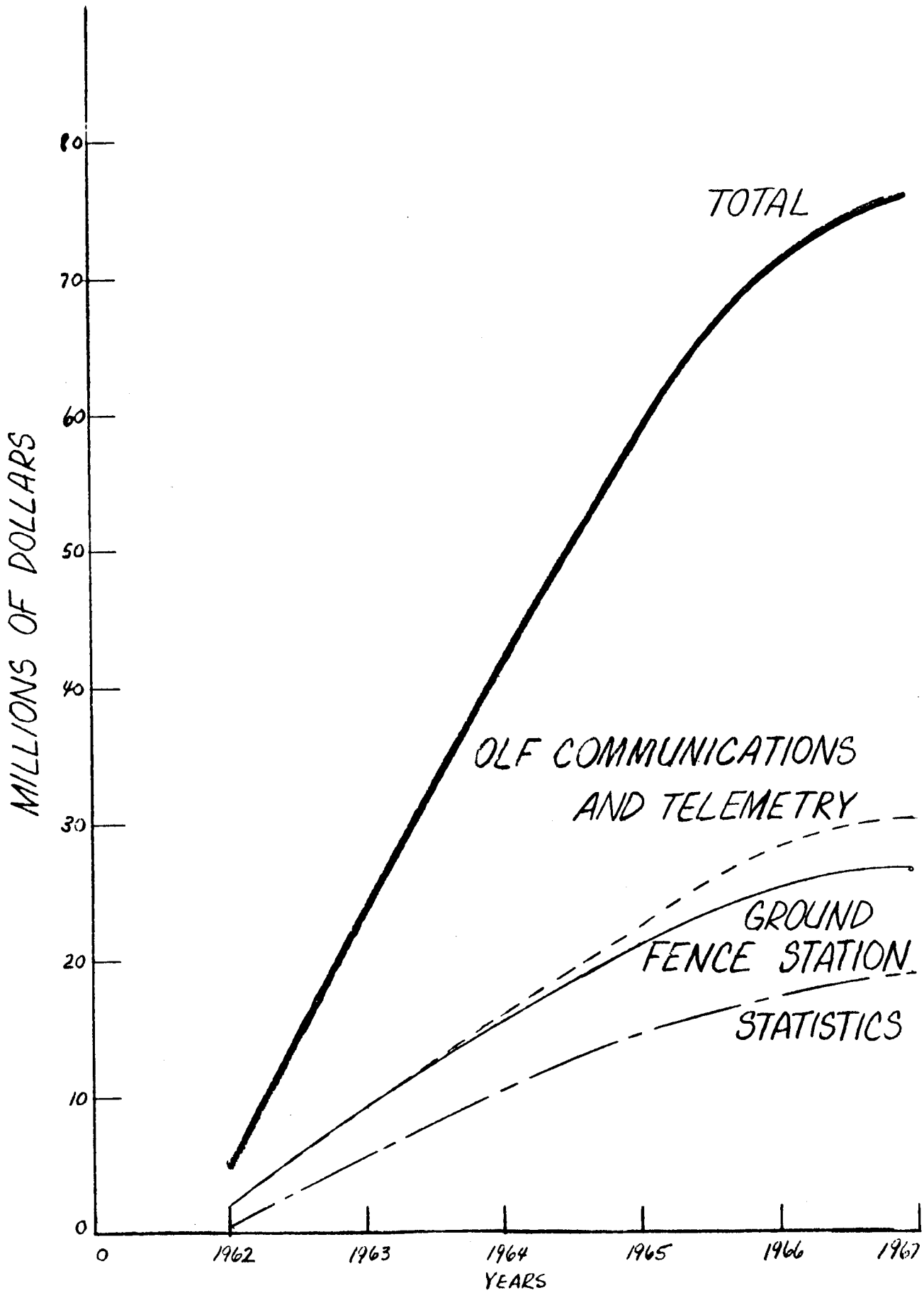



FIGURE 9-5 PROJECTED COSTS

# APPENDIX A

## BASIC NETWORK GEOMETRY

The Orbital Launch Operation communication system has many mobile station requirements. The following oversimplification is given to emphasize the elements of the geometry of the toughest of the communication problems, that of continuity. After the discussion of the five geometric concepts, each concept will be related to the OLO operation.

In figure A-1 a disc rotating at angular velocity $\omega_1$, is shown for consideration. Located on this disc at a discrete location, is user B. External to the disc is source A, fixed with respect to the inertial reference. Source A is tied to the disc with a brush and, B is tied to the periphery of the disc with a fix connection. Figure A-2 shows essentially the same concept as figure A-1 except that the periphery of the disc has been segmented, and user B is connected to each segment. Figure A-3 is similar to figure A-1 except that the brush and the fixed connection have been interchanged and the motion $\omega_1$ is confined to termination A while user B stands still. It is true that the periphery of the disc has been disassociated from the disc and becomes a ring rotating with termination A at rate $\omega_1$.

The concept of figure A-3 may be extended to include other terminations at angular velocity $\omega_1$, termination C for instance, as shown in figure A-4. This may be further extended to include termination C at a different angular velocity from that of termination A, $\omega_2$, and connected to the ring by a brush as shown in figure A-5.

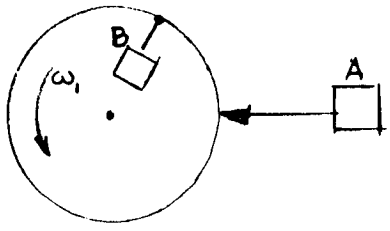



FIGURE A1. GEOMETRIC CONCEPT DISC ROTATING AT ANGULAR VELOCITY $\omega_1$

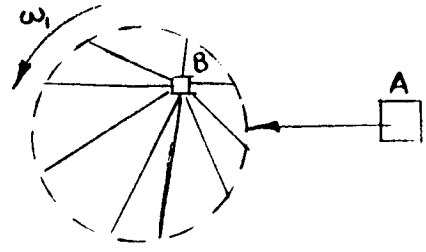



FIGURE A-2 GEOMETRIC CONCEPT SEGMENTED DISC

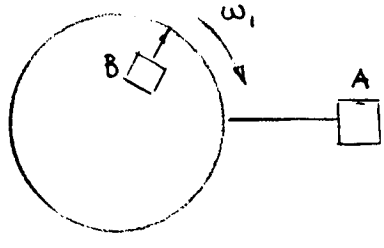



FIGURE A-3 GEOMETRIC CONCEPT INTERCHANGED BRUSH AND FIXED CONNECTIONS

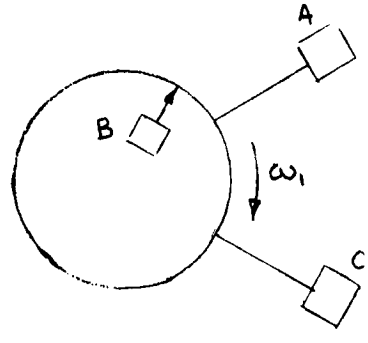



FIGURE A-4 GEOMETRIC CONCEPT ADDITION OF TERMINATION C

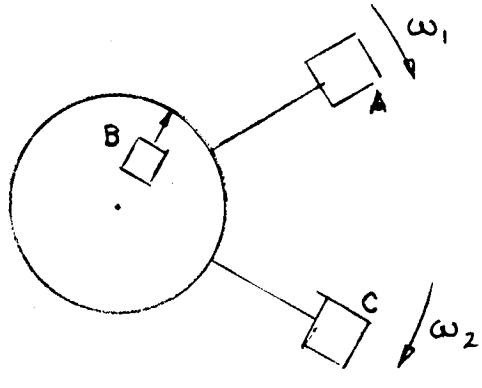



FIGURE A-5 GEOMETRIC CONCEPT. ADDITION OF TERMINATION C AT DIFFERENT ANGULAR VELOCITY

Figures A-1 and A-2 represent a ground station at user B and a mobile space borne station, the OLF, at termination A. The difference between figure A-1 and figure A-2 is the segmenting of the periphery. Figure A-2 represents the present method of a multitude of ground stations, one for each segment connected back to the central control point by radiating connections. Figure A-3 anticipates the possibility of generating a system where the segments are tied together peripherally and rotate with termination A, the OLF. Figure A-4 further extends this to include space borne terminations of angular velocities different from that of the ring.

# APPENDIX B

## TYPES OF CHANNELS

Two types of communication channels find extensive use in our communication systems today. The first type of channel is known as a simplex channel and the second type of channel is a duplex channel.

The simplex type of communication allows traffic to flow only in one direction between the source and the receptor at any given time. Since traffic flow is in one direction only at any given time, supervision of the traffic flow is difficult unless separate supervision channels are provided. For example, if a source is transmitting information, there is no assurance over a simplex system that the receptor has received and understood all of the information until the end of transmission when the direction of transmission is reversed and the receptor is allowed to acknowledge the message. The sender cannot be interrupted for any reason in this system.

The duplex system of communication allows traffic to flow in both directions simultaneously between the source and the receptor. This system then provides a natural supervision system between the ends of the communication channel since either end may interrupt the flow of traffic at any time.

Basically, communication networks may be designed around either of these two basic concepts.

# APPENDIX C

# NET CONTROL CONSIDERATIONS

## C-1. GENERAL

When communication systems expand to more than two terminal stations the factors of interference between stations and system supervision must be considered.

With more than two terminal stations, it becomes necessary to consider the type of communication employed. With the employment of a wire system the communication is bounded and interference between stations is inherently not a problem. If radio links are employed, the only method of eliminating interference between stations is the use of highly directional antennas and multiple channel operation.

Since the communication problem presented in this proposal requires the employment of radio links the discussion which follows will be restricted to this type of communication.

## C-2. SIMPLEX RADIO COMMUNICATION

The simplex type of radio communication will be discussed on the basis of three stations as shown in figure C-1. On any one given frequency at any one given increment of time any one of the stations will be the source and the other stations the receptors. During another increment of time a different transmitter will be the source and the other stations will be the receptors. As mentioned in Appendix B during the

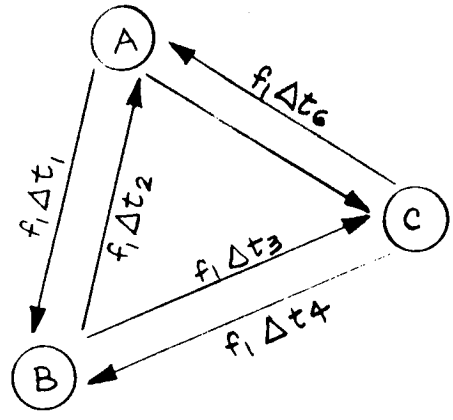



FIGURE C1 SIMPLEX TYPE OF RADIO COMMUNICATIONS

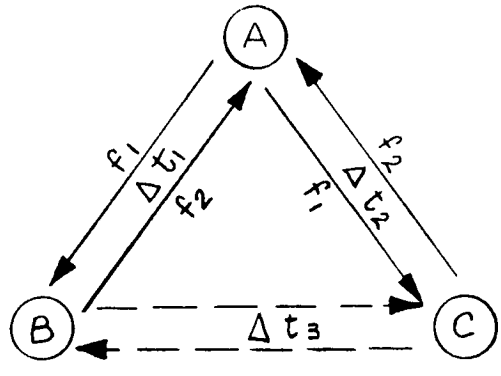



FIGURE C2 DUPLEX TYPE OF RADIO COMMUNICATIONS

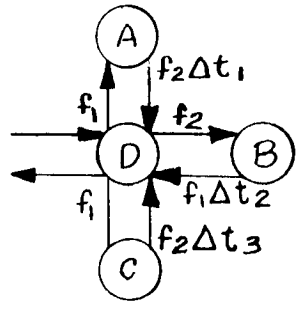



FIGURE C-3 NET CONTROL STATION-METHOD A

increment of transmission of any one given station transmission cannot be interrupted unless supervisory channels are provided.

This type of communication may assume several forms depending upon the length of the increment $\Delta t$. If $\Delta t$ is long the system has the characteristics of point to point simplex communication. As the length of $\Delta t$ becomes shorter the system may have the characteristics of point to point time-shared pulsed type of communication.

## C-3. DUPLEX RADIO COMMUNICATION

The duplex type of radio communication will be discussed on the basis the three stations shown in figure C-2. In this system station A is assigned a transmission frequency of $f_1$ and a reception frequency of $f_2$. Therefore, the stations which are to communicate with station A must be assigned a frequency of transmission of $f_2$ and a frequency of reception of $f_1$. Immediately a problem becomes apparent in that station B cannot communicate with station C on these frequencies since they are both transmitting information on frequency $f_2$ and receiving information on frequency $f_1$.

The solution to this problem in such systems as the high capacity communication system employed by the Navy is the introduction of a net control station.

## C-4. NET CONTROL STATIONS

One method of net control is shown in figure C-3 where station D has the responsibility of receiving all traffic from stations A, B, and C and transmitting all traffic to

A, B, and C. The net control station may also be assigned the responsibility of receiving other communication from other net control stations on other frequencies and distributing this traffic to stations A, B, and C.

A second method of net control is that shown in figure C-4. In this system several frequencies are assigned to the net control station N. The number of frequencies assigned will depend upon the phase of operation and the traffic density. Each station A, B, C or D requests permission for a frequency assignment on a supervisory channel to station N when traffic is to be transmitted. These frequencies may be assigned for any length of time required and then released back to the net control station when the particular transmission is completed.

A second solution to the frequency overlap problem introduced in figure C-2 is shown in figure C-5. Each station has one transmitter and several receivers. The number of receivers required depends upon the number of communication links. Some type of band filtering must be employed to allow operation of the transmitters and receivers simultaneously for full duplex operation. This filtering may be done by electronic filtering, transmitted and received wave polarization, transmission and reception directivity or large spacing between the transmitter and receiver. The method usually employed in restricted space is an electronic type of filtering. Stations A, B, C, and D are operated on frequencies $f_{1a}$, $f_{1b}$, $f_{2a}$, and $f_{3a}$.

The method of employment of the filters $f_1$, $f_2$, and $f_3$ is shown in figure C-6. The bands $f_1$, $f_2$, and $f_3$ are any band of non-overlapping frequencies chosen for the system. Frequency $f_{1a}$ is any discreet frequency in band $f_1$. Frequency $f_{1b}$ is another discreet frequency in band $f_1$. Frequency $f_{2a}$ is some discreet frequency in band $f_2$. This type of assignment

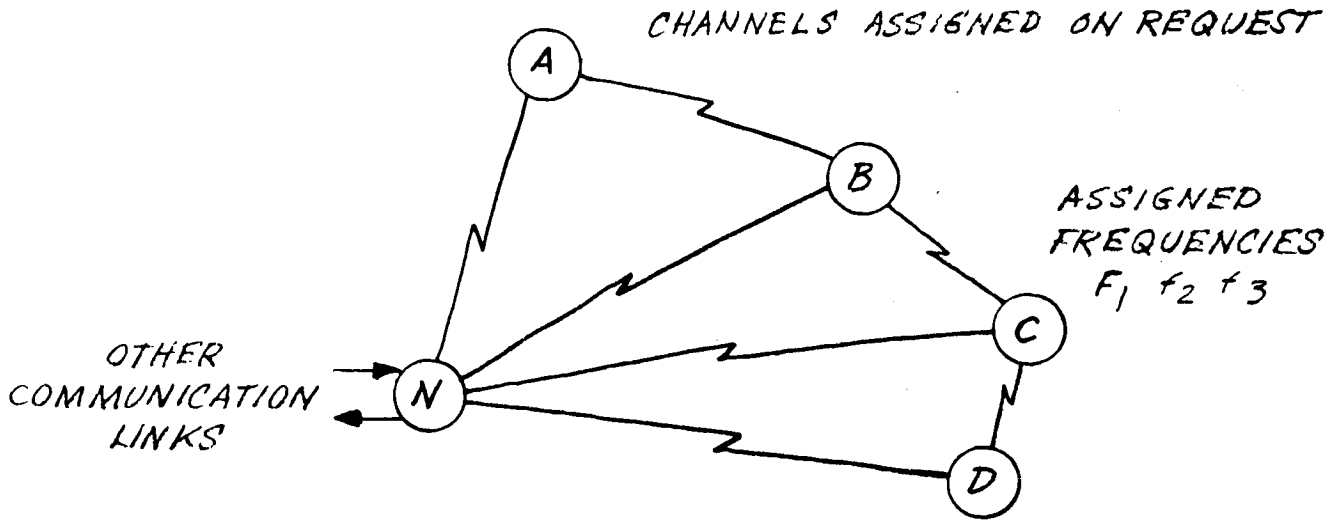



FIGURE C4 NET CONTROL STATION METHOD B

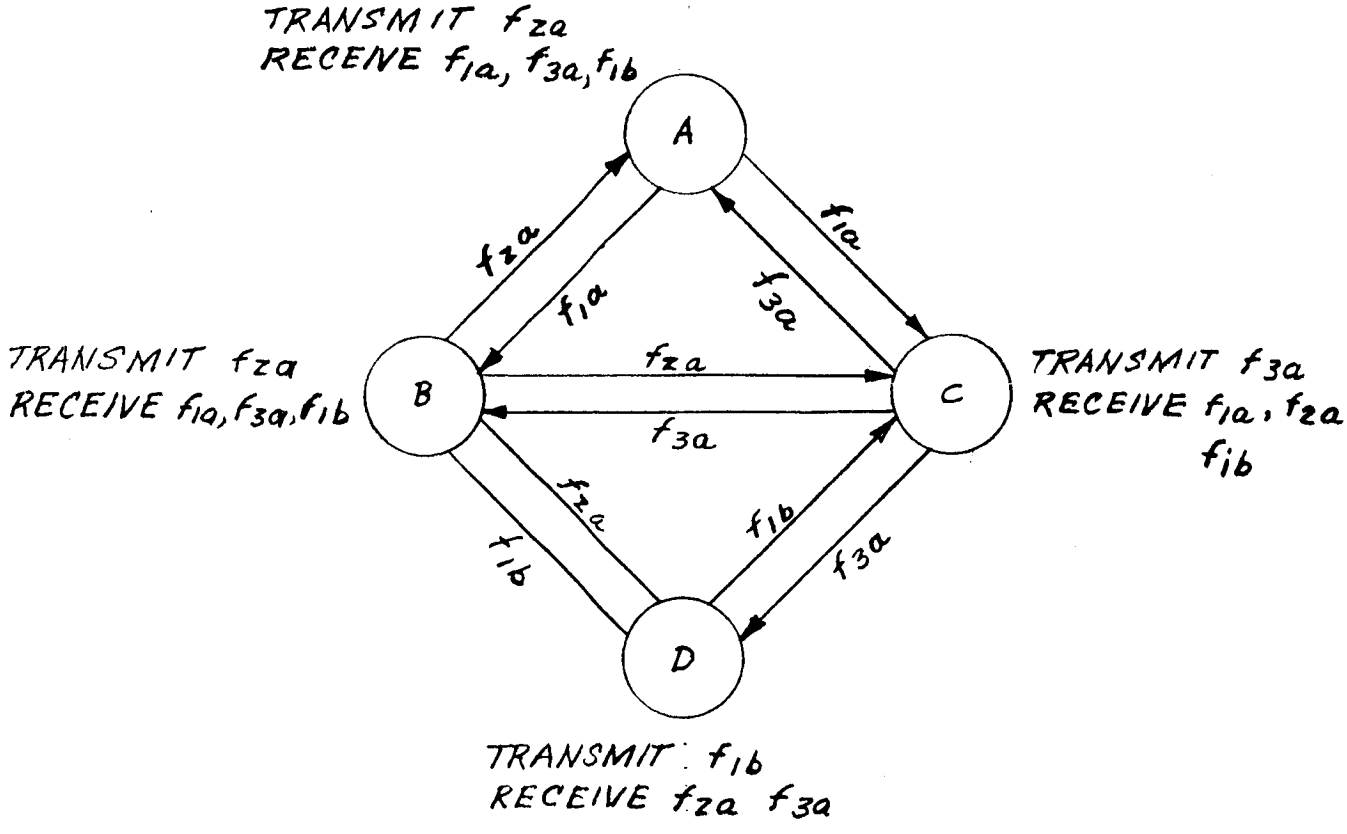



FIGURE C-5 NET CONTROL STATION METHOD C

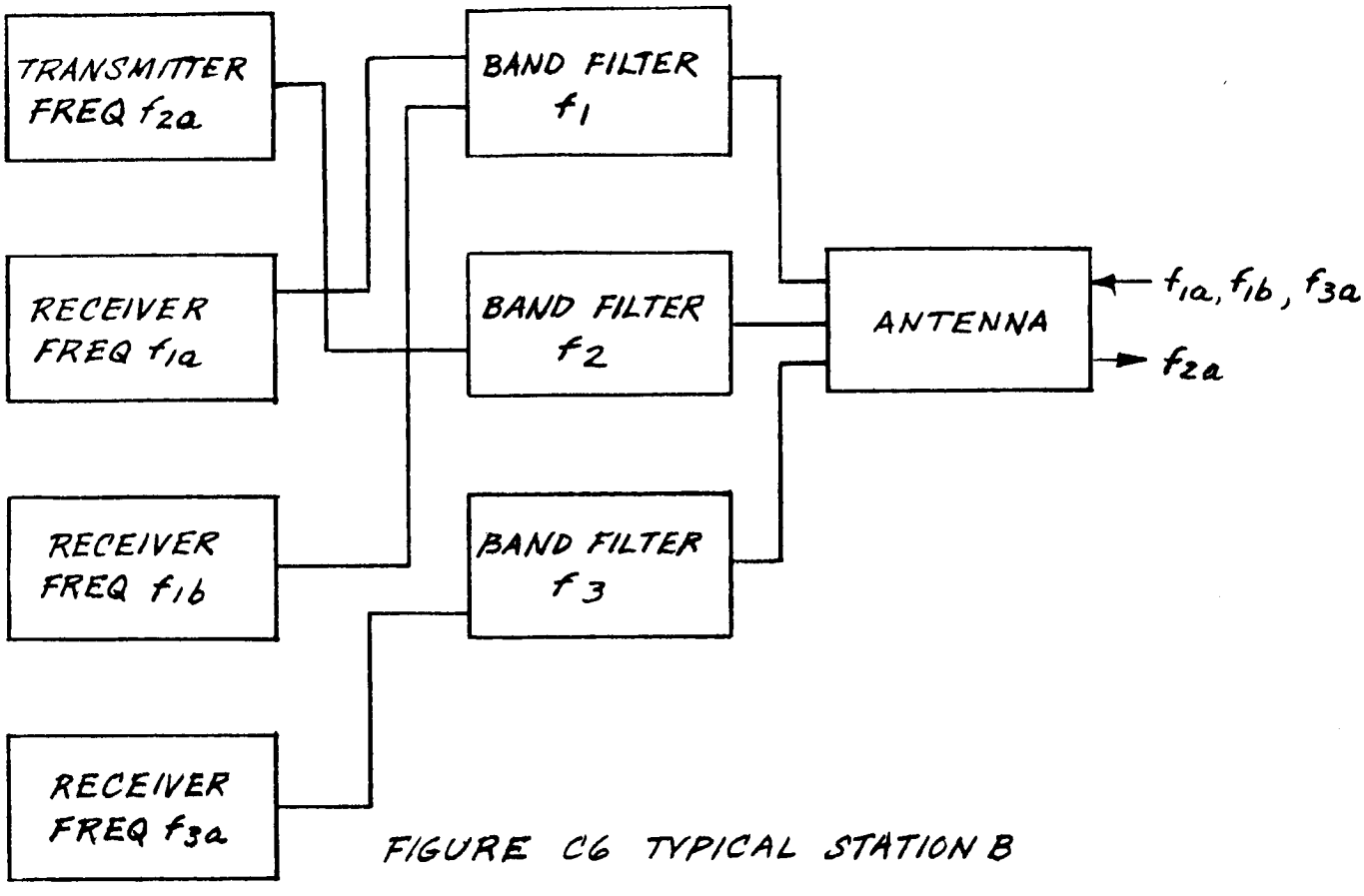



FIGURE C6 TYPICAL STATION B

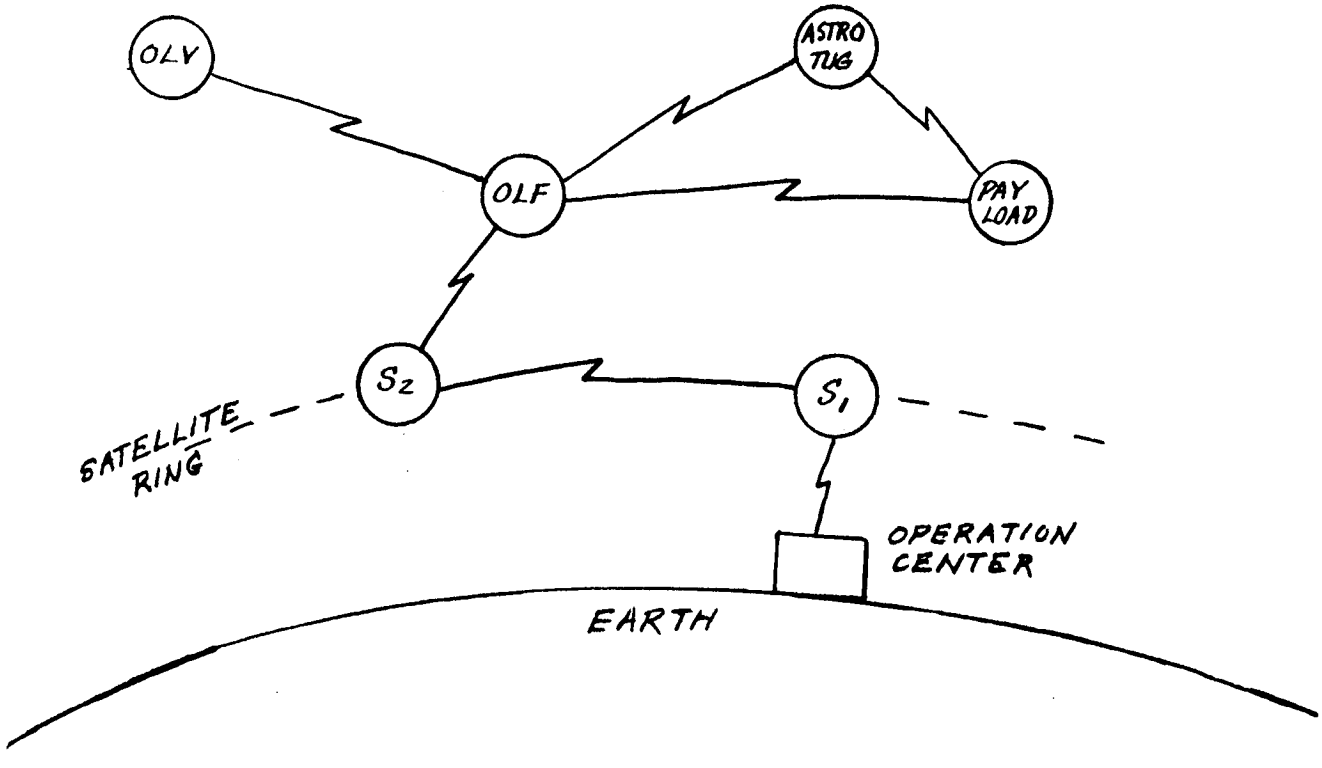



FIGURE C-7 FREQUENCY PLAN FOR OLF COMMUNICATION LINK

is carried to completion for the station under consideration. If the band filters are designed to be able to fulfill the transmit or receive requirements, complete patching flexibility may be achieved. The only difference between station B and station C in figure C-5 would be the interconnections between the transmitters, receivers and band filters and the frequencies of the transmitters and receivers.

Such a frequency plan provides for the OLF communication links as shown in figure C-7. Although there is continuous motion of the OLF complex with respect to the operations center, there is always a possible relay path between the operations center and any vehicle in space through the satellite ring. Other possible paths may be established upon demand. The paths suggested in figure C-7 are a few of the possible requirements.

A sequence of possible required paths are shown in figure C-8A through F. These examples are by no means restrictive. Many other possible combinations may be suggested by the operational requirements. Figure C-8A-D shows a possible required system just after the astro tug and OLF become operational. The payload is in communication with the ground just after launch in figure C-8a. If operation remains overhead it may communicate with the ground until it is in range of the OLF and then the OLF directly. (An example of this operation is a sounding rocket.) Figure C-8C illustrates the inclusion of the payload into the OLF to prepare for a docking operation of figure C-8D. Figure C-8E illustrates a possible communication complex later in the program. Note that figure C-8E and figure C-8F illustrate the flexibility of the system. In this case a payload is launched at some distance from the operations center. Communications may then be required with the OLF and operations center immediately after launch. This becomes possible in figure C-8F through the satellite relay ring.

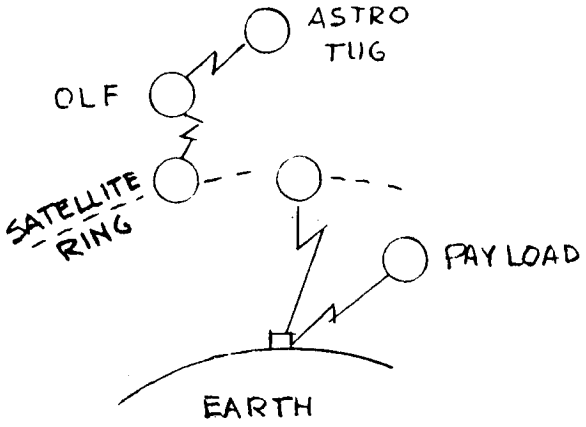



FIGURE 8a

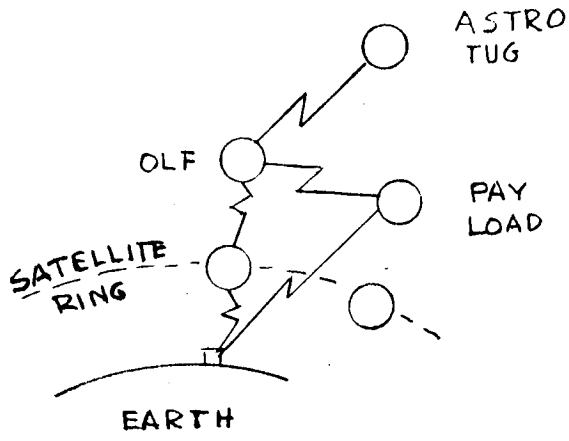



FIGURE 8b

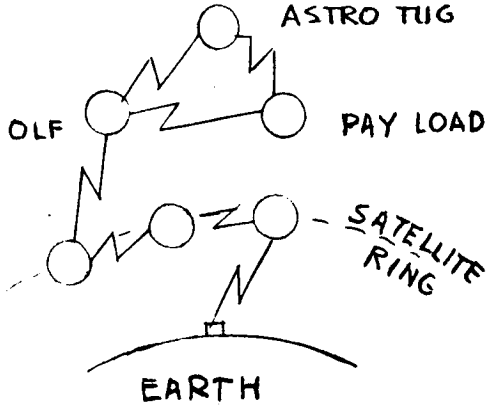



FIGURE 8c

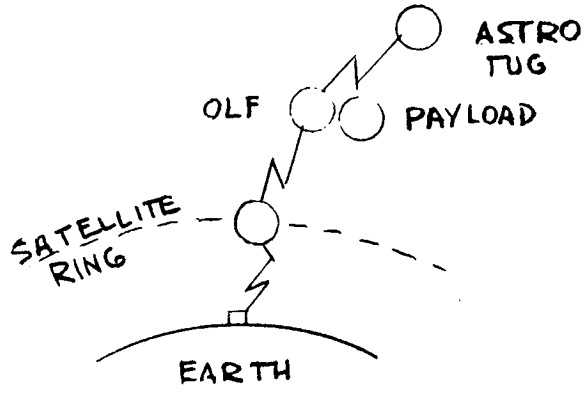



FIGURE 8d

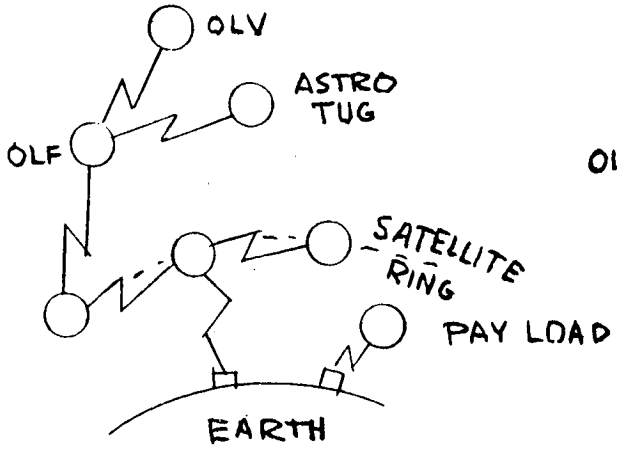



FIGURE 8e

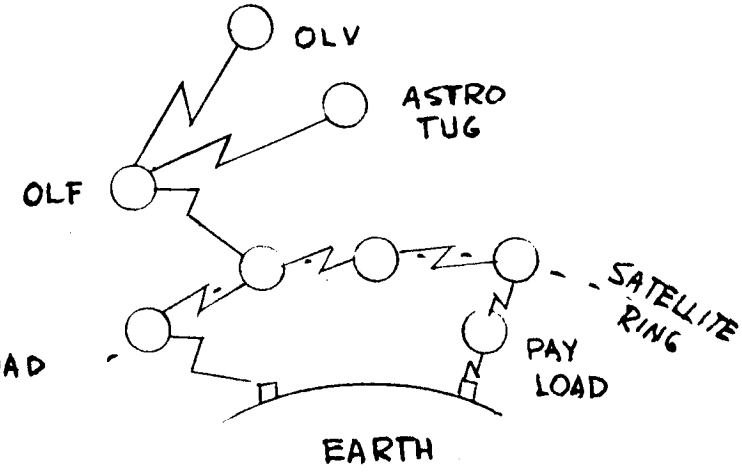



FIGURE 8f

FIGURE C-8 SEQUENCE OF POSSIBLE COMMUNICATION LINKS

# APPENDIX D

## THE FENCE

In the system proposed, the fence has the assignment of receiving and transmitting to and from the satellite ring and then relaying this information to the center of operations. The first part of this assignment is limited in capability (bandwidth and signal-to-noise ratio) within all practical limits by the power or weight of the space borne equipment.

The immediate solution of the second half, that of utilizing HF radio along the fence, introduces a limitation on capability that is somewhat independent of equipment. The assignment of wide channels in the HF spectrum and the vagaries of the ionosphere as a transmission medium, have a considerable effect on what may be done by means of HF radio. None the less the HF solution has been pursued. Overall comparisons may then be made with other means. Cables, or ground-to-ground communication via random or 24 hour satellites should be reexamined as candidates for improvement of the system. Unfortunately time has not permitted the evaluation of these aspects of the problem and they must be relegated to the list of items for further study.

Figure D-1 is a block diagram of a typical fence relay station. Each station may be divided into three sections, the equipment to and from the ring, the equipment for up fence operation and the equipment for down fence operation. The last two portions are identical. All three of these sections are never used simultaneously. The dotted arrows in the

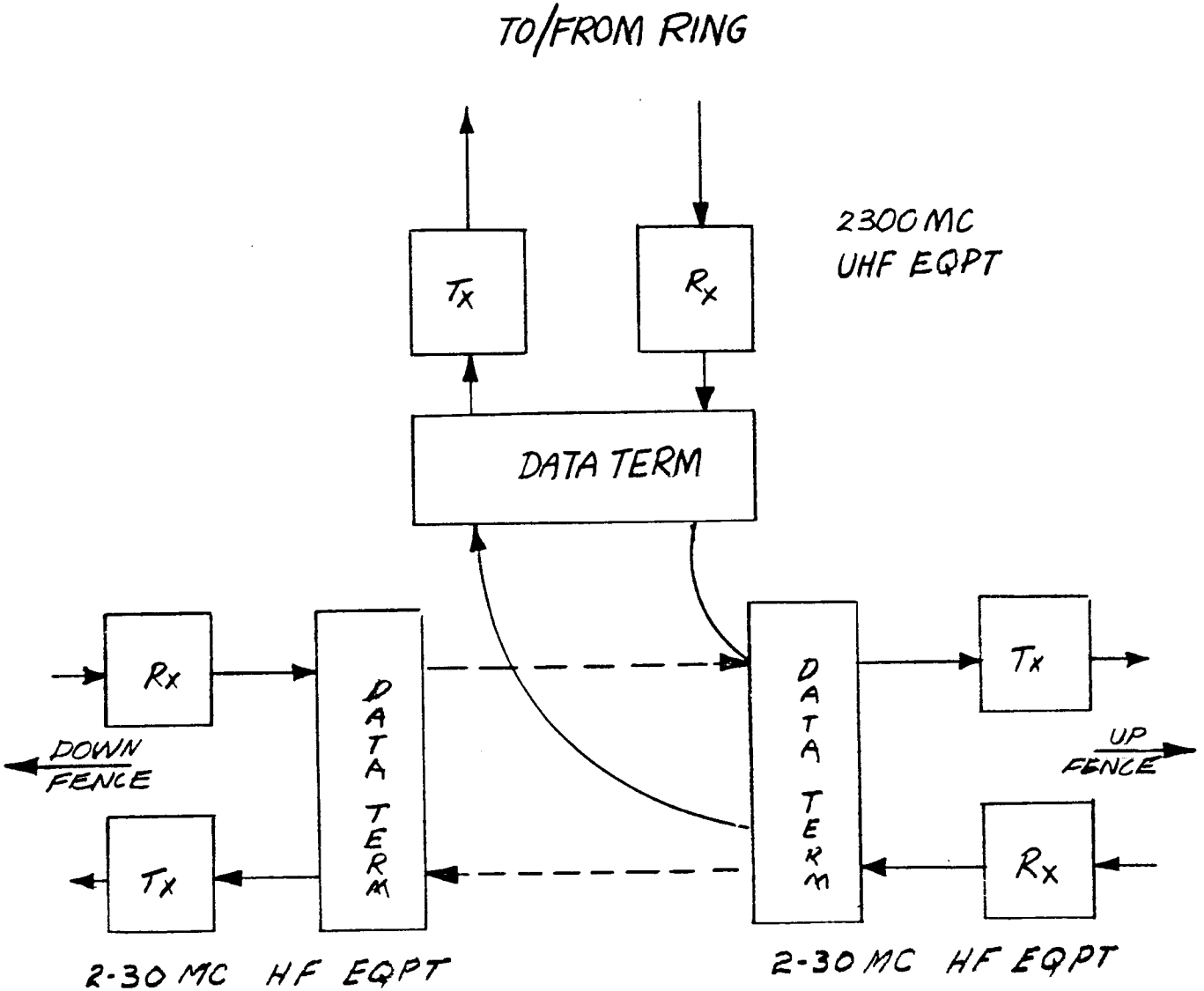



FIGURE D-1 FENCE RELAY STATION

center of the diagram indicate the connections that are used when the station is used as an HF relay only with the UHF pick up from the ring being done farther down fence. These sections are made up of transmitters, receivers and data terminals. Voice paths through the equipment have been omitted for simplicity. Further details of each section will be given after a short discussion of the use of this chain of station.

The path from ring to Cape includes one station utilizing its UHF equipment to and from the ring plus one or more stations utilizing both halves of the HF equipment up and down fence as a relay. The most northerly and most southerly station require only one half the HF equipment. The fence stations shown in figure 4-3 of the main text were selected to coincide with stations used for other programs in so far as possible. They are closer together than necessary but this is an advantage since it provides some built in alternate routes so important to overall reliability. Each hop is on the order of 750 statute miles and therefore there are many combinations that might be used in returning to the Cape. Although so far as managing the chain is concerned, it would be simple to always relay through the next northerly station, however, this would not be the most desirable from a frequency assignment basis. Single hops should be utilized when propagation is good and multiple hops when propagation is bad. In this regard, it is interesting to note that north-south paths are to be preferred to east-west paths for reliability on HF circuits.

Returning to further details of the stations, figure D-2 shows the UHF section. It includes a tracking antenna, a diplexer, a preamplifier, a receiver, a data terminal and a transmitter. The antenna gain is selected at 48 db in order to reduce the equipment in the satellite to a minimum weight.

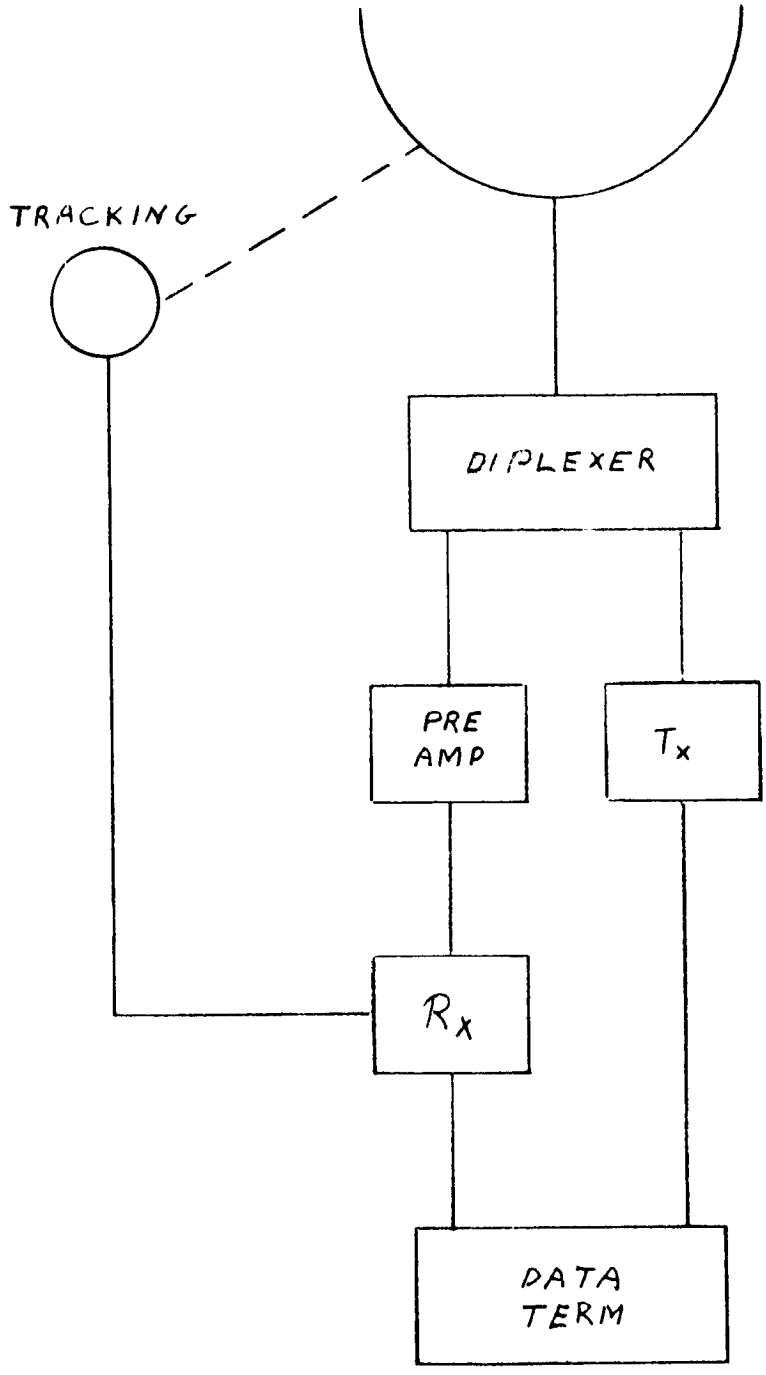



FIGURE D-2 2300 MC SECTION OF FENCE RELAY STATION

This gives a 0.5 degree beam width which requires accurate tracking. The receiver is of advanced design with a $200^{o}$ k noise temperature. The transmitter is quide modest, requiring only ten watts. The data terminal has been included since it is believed that the most flexible system is that in which the data is demodulated to binary form and coverted to multi-channel binary form at a slower individual rate to meet the requirements of propagation via the HF medium. Even if the proper composite form already exists as transmitted from the space craft, relaying through several HF hops requires re-generation so that terminal equipment is required at each station. In addition, there are probably some channels of data that have higher priorities than others contained in the composite signal from the UHF circuit. Demodulation would be required to be sure to forward the priority portions even under conditions of reduced capacity up the fence.

The HF section of the fence station consists of several HF systems in parallel each having a capability of 16-kc as shown in figure D-3. Separated transmitting and receiving antennas are shown. Three transmitters are combined on two single antennas. However, the filters are almost the same price as large directive antennas and if land is no problem, separate antennas might be used for each transmitter. The transmitter line up includes the antenna, the filter, the power amplifier, the exciter, and the data terminal. The antenna suggested is a vertical logarithmically periodic antenna covering 2-30 mc having a gain of 7 db over a quarter wave monopole. A bandpass filter tuned to the transmitting frequency having an attenuation of 20 db for the signal from the adjacent transmitter is required. The PA's are manually tuned 10 kw PEP 2-30 mc linear amplifiers. The exciters are manually adjustable in 1-kc steps from 2-30 mc. The data

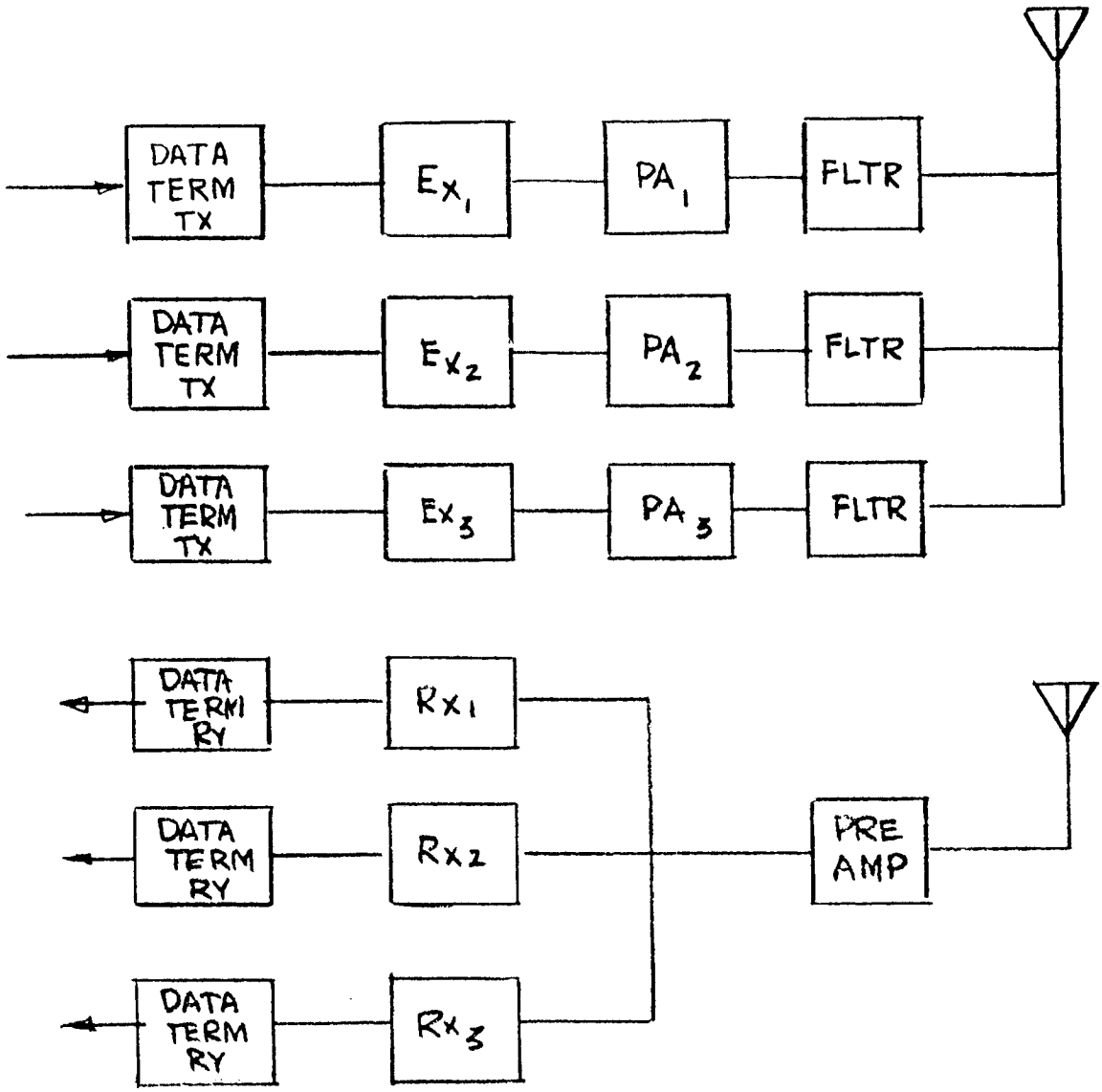



FIGURE D-3 HALF OF THE HF SECTION OF THE FENCE RELAY STATION

terminals are four phase state, frequency multiplexed, bit synchronous equipment giving about 1 bit per second per cycle of bandwidth (Collins Kineplex). Figure D-3 shows 3 x 16-kc or 48-kc of bandwidth. The limitation here is not equipment but 48-kc appears to be near an upper limit of the possible frequency assignments. Of course such a system could be implemented in steps starting with a single 16-kc band or less.

Figure D-4 is a table showing fence costs, $26,000,000 for ten years of operation.

| ITEM<br>COST $ | BLDG<br>100K | HF ANT<br>7 db 2-30 mc<br>20K | FILTER<br>20K | PA<br>10 kw<br>15K | EX<br>2-30 mc<br>10K | Pre Amp<br>2-30 mc<br>10K | RX<br>2-30 mc<br>10K | TERM<br>KPX<br>10/B/S<br>(BW = B/S) | 2300 mc<br>RF<br>200K | DATA<br>TERM<br>500K |
|---|---|---|---|---|---|---|---|---|---|---|
| STN | | | | | | | | | | |
| 1 | 1 | 2 | 3 | 3 | 3 | 1 | 3 | 48K | 1 | 1 |
| 2 | 1 | 4 | 6 | 6 | 6 | 2 | 6 | 96K | 1 | 1 |
| 3 | 1 | 4 | 6 | 6 | 6 | 2 | 6 | 96K | 1 | 1 |
| 4 | 1 | 4 | 6 | 6 | 6 | 2 | 6 | 96K | 1 | 1 |
| 5 | 1 | 4 | 6 | 6 | 6 | 2 | 6 | 96K | 1 | 1 |
| 6 | 1 | 2 | 3 | 3 | 3 | 1 | 3 | 48K | 1 | 1 |
| TOTAL | 6 | 20 | 30 | 30 | 30 | 10 | 30 | 480K | 6 | 6 |
| | 600K | 400K | 600K | 450K | 300K | 100K | 300K | 4800K | 1200K | 3000K |



Total radio equipment and buildings 11,750K

PWR 88K/STN/yr at 10¢ KWH 100 KW

Personnel 75K/STN/yr at 15K/M x 5 min

Repairs 1,300K/yr

For ten years - 26 Million

Figure D-4 - Fence Costs for 48 KC Bandwidth

# APPENDIX E

# THE SATELLITE RING

## E-1. GENERAL

A ring of satellites as shown in figure E-1 must have certain features to allow flexible operation. This ring may be compared to a microwave relay link across country. The distance between stations becomes greater since the line of sight range is greater. A gap between satellites No. 1 and No. 12 in the ring prevents closed loop oscillations in the system. There is no break in coverage however since satellite No. 1 and satellite No. 12 overlap each other in coverage. Several entrances will be provided at each satellite for communication from the ground OLF, S/C, and payloads.

## E-2. FREQUENCY PLAN

Employment of the previously mentioned principles will result in the development of a frequency plan. This plan is based upon multi-band filters for the OLF, S/C, astrotug and satellite ring. One of the many possible frequency plans is shown in figure E-2. This figure indicates the number of frequency bands which would be required. The specific system under final consideration will determine the number of band-filters, transmitters, receivers etc. that are necessary for the system. In figure E-2 $f_1$, $f_2$, etc. indicate the number of bands required in the communication plan. These bands determine the number of band-filters required.

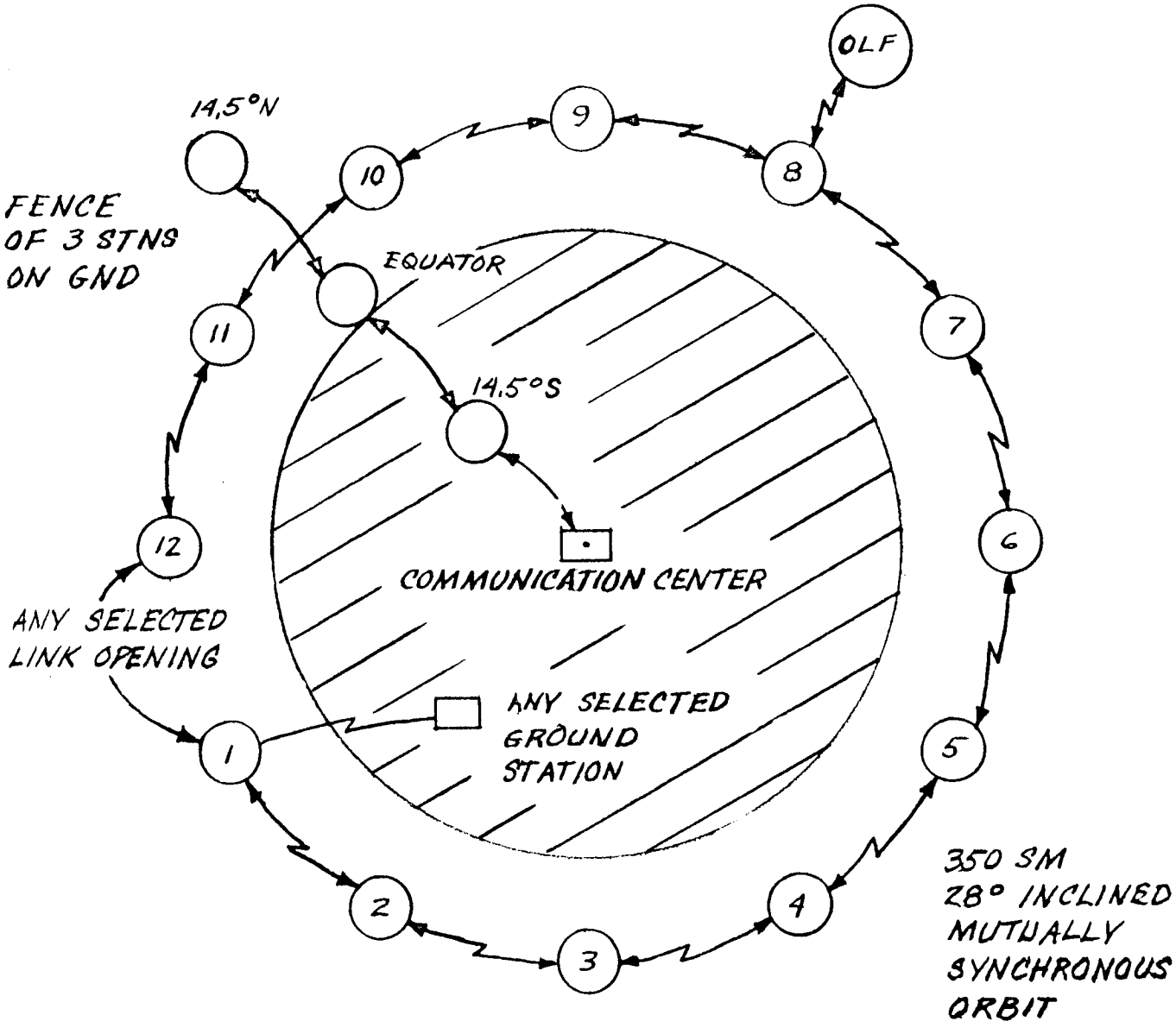



FIGURE E-1 OLF- SATELLITE RING-
GROUND FENCE- CONCEPT

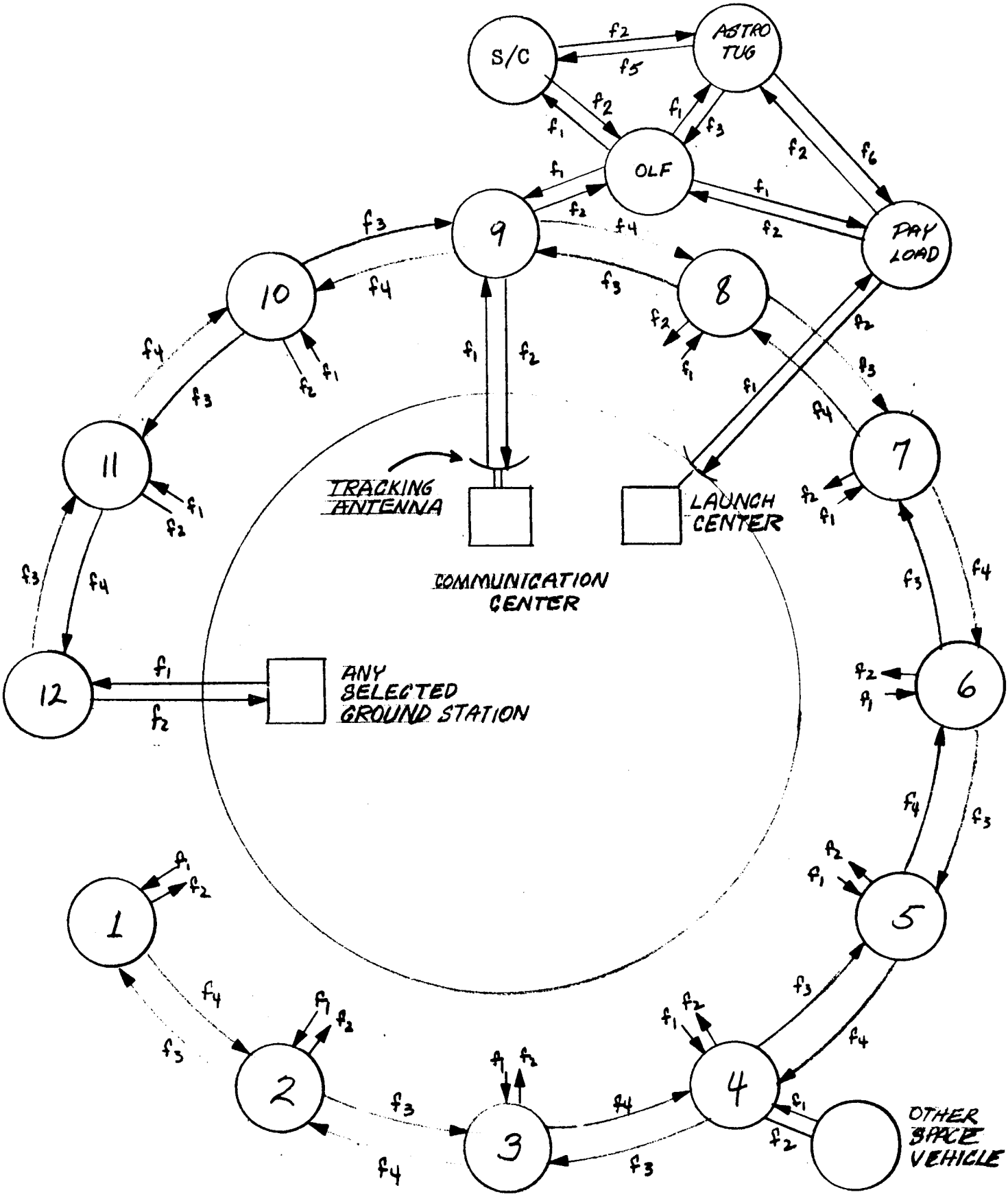



FIGURE E-2 RING FREQUENCY PLAN

Calculations of candidate hardware for the satellite ring are presented in the following paragraphs. They are based on a 40 db signal-to-noise ratio at each station or a minimum overall signal-to-noise ratio of 30 db. This provides excellent error rates and may be on the conservative side. Any conservatism may be converted directly into added bandwidth at double the bandwidth for each 3 db of signal-to-noise. The 200-kc is I-F bandwidth and for low deviation F-M system must be divided by about 3. Call this 75 kc of base bandwidth with 30 db signal-to-noise ratio from OLF to fence.

The calculations are based on the following concepts:

a. All satellite receivers and transmitters are of elementary variety. Receivers are crystal or traveling-wave tube (TWT) type. Power of the transmitters is held to a low level. Two watts at 2.2 kmc is attainable with all solid state devices. Powers above this require a tube amplifier. Higher solid state powers may be obtainable at the time of unit development.

b. The size of the antenna for the satellite to OLF link is restricted by the orbit of the OLF with respect to the satellite.

c. The size of the satellite antenna is restricted by the degree of orientation accuracy of the satellite.

d. Parametric receivers are allowed in the OLF and a maser receiver is allowed on the ground.

e. An orientable antenna is necessary on the OLF, S/C and astrotug. (mechanical or electronic orientation).

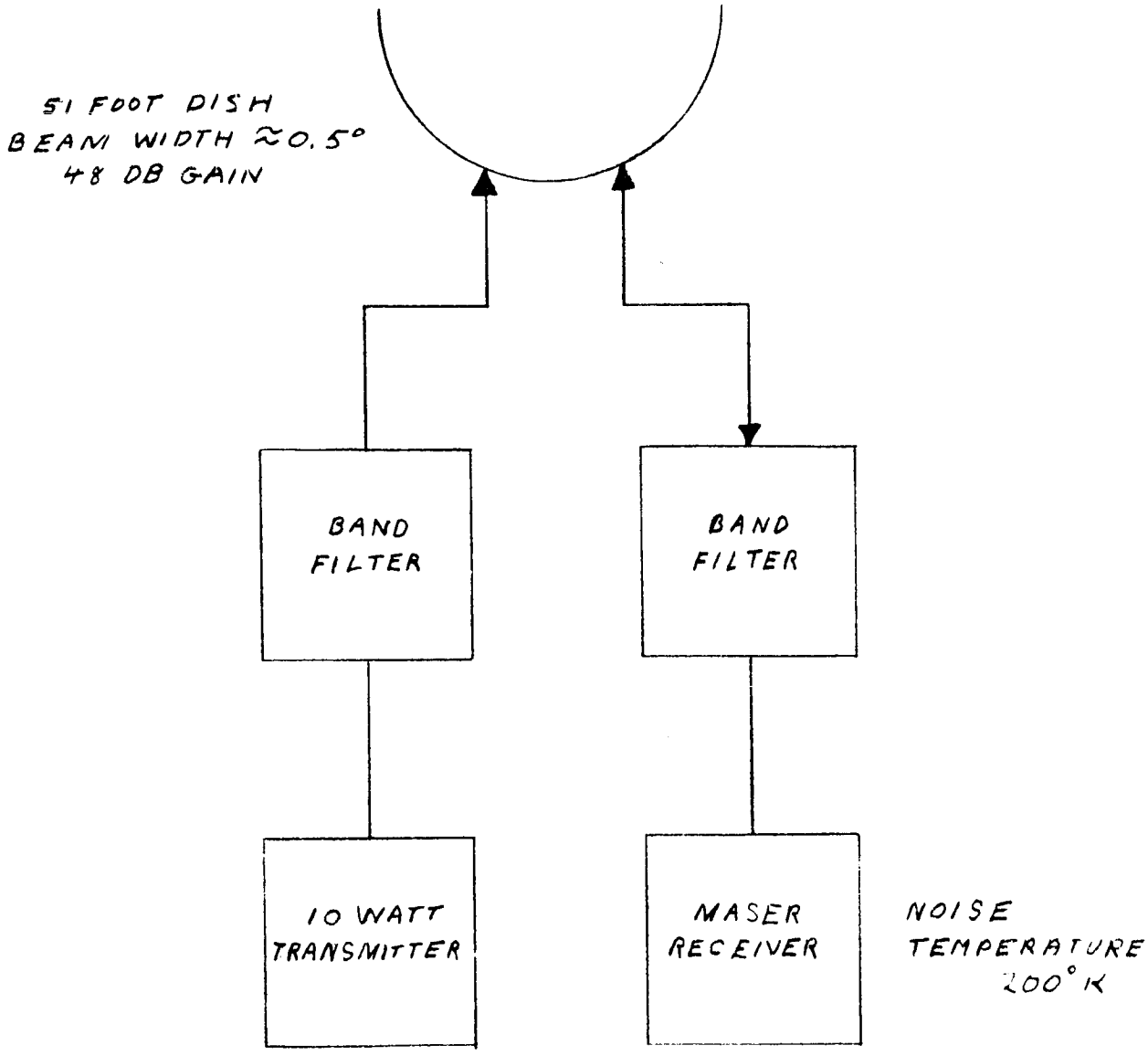



FIGURE E-3 GROUND STATION

Six sets of calculations follow for the six paths involved. Following these calculations is a summary of costs, weights and power requirements for the communication satellites.

## E-3. 200-KC SYSTEM

a. Ground Station Range (350 to 1450 miles). Refer to figure E-3.

S/N 40 db = $10^4$

Bandwidth $2 \times 10^5$

Receiver Noise Temperature 200°K

$\lambda$ = 5 inches (2300 MC)

Gt = Gain of Transmitting Antenna 3 db (GAIN = 2)

Gr = Gain of Receiving Antenna

Pt = Power of Transmitter in Watts (2 WATTS)

SATELLITE TRANSMIT EARTH RECEIVE

$$Gr = \frac{S/N \; B \; 16\pi^2 \; R^2 \; k \; Tr}{\lambda^2 \; Gt \; Pt}$$

$$Gr = \frac{10^4 \times 2 \times 10^5 \times 16\pi^2 \; (1450)^2 \times (5280)^2 \times 1.38 \times 10^{-23} \times 200}{(5/12)^2 \times 2 \times 2}$$

$Gr = 7.35 \times 10^4$ or $Gr = 48.66$ db

$Gr = \frac{6.3A}{\lambda^2}$ $\quad A = \frac{\pi D^2}{4}$

$A = \frac{Gr\ \lambda^2}{6.3} = \frac{7.35 \times 10^4}{6.3} (5/12)^2$ $\quad D^2 = \frac{4A}{\pi}$

$= \frac{7.35 \times 10^4 \times 25}{6.3 \times 144}$ $\quad = \frac{4}{\pi} \times 2.03 \times 10^3$

$= 2.03 \times 10^3$ $\quad = 0.259 \times 10^4$

$D = 0.51 \times 10^2$

$D = 51$ ft DISH

b. Earth Transmit - Satellite Receive Ground Station Range (350 to 1450 miles)

S/N 40 DB = $10^4$

Bandwidth $2 \times 10^5$

Receiver noise temperature 870°K (crystal receiver or twt)

$\lambda = 5$ inches (2300 MC)

Gt = Transmitter antenna gain $7.35 \times 10^4$

Gr = Gain of receiving antenna (3db) Gr = 2

Pt = Power of transmitter in watts

$$Pt = \frac{B\ S/N \times 16\pi^2\ R^2\ k\ Tr}{Gr\ Gt\ \lambda^2}$$

$$= \frac{2 \times 10^5 \times 10^4 \times 16\pi^2 \times (1450)^2 \times (5280)^2 \times 1.38 \times 10^{-23} \times 870}{2 \times 7.35 \times 10^4 \times (5/12)^2}$$

$= 6.23$ watts

Pt assigned 10 watts

c. OLF → Satellite

OLF → Satellite Range Maximum 2400 miles. OLF has tracking antenna.

S/N 40 db = $10^4$

Bandwidth 2 x $10^5$

Receiver noise temperature 870°K

$\lambda$ = 5 inches (2300 MC)

Gt = Transmitting antenna gain 8 ft dish 32.6 db 1825 gain

Gr = Receiving antenna gain = 124 (20.94 db) 2 ft dish

$$Pt = \frac{B \times S/N \times 16\pi^2 \times R^2 \times k\ Tr}{Gr\ Gt\ \lambda^2}$$

$$= \frac{2\times10^5 \times 10^4 \times 16\pi^2 \times (2400)^2 \times (5280)^2 \times 1.38 \times 10^{-23} \times 870}{124 \times (5/12)^2 \times 1825}$$

= 7.95 watts

Use 8 watt transmitter on OLF

d. Satellite → OLF

S/N 40 db = $10^4$

Bandwidth 2 x $10^5$

Range 2400 MILES

B = 200KC

$\lambda$ = 5 inches (2300 MC)

Gt = Gain of receiving antenna 8 ft dish (32.6 db 1825 gain)

Gr = Gain of Transmitting antenna 2 ft dish (20.93 db) 124 gain

Tr = 340°K (Parametric Receiver)

$$Pt = \frac{B \times S/N \times 16\pi^2 \times R^2 \; k \; Tr}{Gr \; Gt \; \lambda^2}$$

$$= \frac{2x10^5 x10^4 x16\pi^2 x(2400)^2 x(5280)^2 x1.38x10^{-23} x340}{124 \times 1825 \; (5/12)^2}$$

Pt = 3.1 watts

use 4 watt transmitter

e. Satellite to Satellite

S/N 40 db = $10^4$

Bandwidth 2 x $10^5$

Range 2400 miles

k = 1.38 x $10^{-23}$

Tr = 870°K (crystal or twt receiver)

$\lambda$ = 5 inches (2300 MC)

Gt = Gain of transmitting antenna 8 ft dish (32.6db) 1825 gain

Gr = Gain of receiving antenna 8 ft dish (32.6db) 1825 gain

Pt = Power of transmitter watts

$$Pt = \frac{13 \; S/N \; 16\pi^2 \; R^2 \; k \; Tr}{Gr \; Gt \; \lambda^2}$$

$$Pt = \frac{2x10^5 x10^4 x1.579x10^2 x1.60\%x10^{14} x1.38x10^{-23} x.87x10^3 x1.44x10^2}{(1.6[illegible])^2 \times 2.5 \times 10^7}$$

Pt = 1.04 watts

$$Gt = \frac{6.3A}{\lambda^2} = \frac{6.3(4)^2\,\pi}{(5/12)^2}$$

$$= 1825$$

$$= 32.6 \text{ db}$$

$$\text{Beam } \alpha = \frac{60\lambda}{D} = \frac{60 \times 5}{8 \times 12}$$

$$= \frac{120}{72} = 1.67$$

$$= 3.340$$

f. OLF to Astrotug

Range in miles desirable = 300

$$Gr\ Gt = \frac{B\ S/N \times 16\pi^2 \times R^2\ k\ Tr}{\lambda^2}$$

$$= \frac{2\text{x}10^5\text{x}10^4\text{x}16\ ^2(300)^2 \times (5280)^2\text{x}1.38\text{x}10^{-23}\text{x}870}{(5/12)^2}$$

$$= \frac{2\text{x}10^9\text{x}1.579\text{x}10^2\text{x}2.51\text{x}10^{12}\text{x}1.38\text{x}10^{-23}\text{x}.87\text{x}16^3\text{x}1.44\text{x}10^2}{25}$$

$$= \frac{2\text{x}1.579\text{x}2.51\text{x}1.38\text{x}.87\text{x}1.44\text{x}10^3\text{x}4}{.25}$$

$$= 55 \times 10^3$$

$$= 5.5 \times 10^4$$

$$Gr = Gt = 2.34 \times 10^2$$

$$= 234$$

$$Gt = \frac{6.3\pi D^2}{\lambda^2 4}$$

$$4Gt\ \lambda^2 = 6.3\pi D^2$$

$$D^2 = \frac{4Gt\ \lambda^2}{6.3\pi}$$

$$D^2 = \frac{4 \times 234 \times (5/12)^2}{6.3}$$

$$D^2 = \frac{4 \times 234 \times 25}{6.3 \times 144} = 8.22$$

D = 2.87 ft dish with Beam

E-4. SATELLITE EQUIPMENT COSTS

| | | Wt (lbs) | Watts Input |
|---|---|---|---|
| 4 Receivers | 30,000 | 32 | 40 |
| 2 1 watt transmitters | 24,000 | 8 | 60 |
| 1 2 watt transmitter | 12,000 | 4 | 35 |
| 1 4 watt transmitter | 15,000 | 5 | 50 |
| 1 Duplex coupler | 15,000 | 3 | 5 |
| 8 Band pass filters | 24,000 | 8 | - |
| 2 8 ft dishes | 6,000 | 60 | - |
| 1 2 ft dish | 3,000 | 10 | - |
| 1 Slot antenna | 800 | 5 | - |
| | $129,800 | 135 | 190 |
| Power Supply System Batteries and Solar Cell | 90,000 | 120 | |
| Control Equipment, Electronic Control Equipment | 12,000 | 20<br>20 | |
| | 231,800 | 295 | |
| Frame | 100,000 | 100 | |
| | 331,800 | 395 | |
| Development | 4,000,000 | | |
| Make 12 ft units | 3,600,000 | | |
| 12 spares, test, train | 3,600,000 | | |
| | 11,200,000 | | |

# APPENDIX F

# CANDIDATE OLF TELEMETRY SYSTEM

## F-1. CANDIDATE SYSTEMS

### a. General

The initial requirements placed on the OLF communications system were discussed in Section VIII. In the selection of a system with the capability of meeting these requirements, a number of factors bearing on each requirement were considered. These factors include accuracy, reliability, checkout, repair, power requirements, weight, size, and interchangeability. It must be recognized that these relevant factors may vary in importance depending on the mission requirements of the system. The following discussion of condidate systems is based in part on the relative weighting of the system characteristics discussed.

In view of the fact that an analog system, a digital system, and an analog-digital combination provide a representative sampling of the numerous special combinations of telemeters in use, these systems were subjected to close evaluation. Specifically, a PAM/FM/FM, a PCM/FM, and a hybrid system employing PAM/FM, PCM/FM, and FM/FM were chosen for evaluation.

b. Evaluation of Candidate System. A program-orientated evaluation was performed on the candidate systems to determine OLO mission efficacy. The results of this evaluation are shown in figure F-1. The system rated best in each category is scored with a 1. If identical abilities exist for more

than one system, each is scored with a 1. The implications of the scoring in figure F-1 are treated in the following paragraphs.

(1) Modulation (carrier). The FM modulated carrier was chosen because of the FM signal-to-noise improvement and because of compatibility with projected ground facilities and other OLO communications.

FIGURE F-1

Comparison of Candidate Systems

| | PCM/FM | PAM/FM/FM | HYBRID (PCM/FM/FM) (PAM/FM/FM) (FM/FM) |
|---|---|---|---|
| Optimum carrier modulation | 1 | 1 | 1 |
| Data format | 1 | | 1 |
| Bandwidth (minimum) | 1 | | |
| Transmitter power (minimum) | 1 | | |
| Data storage | 1 | | |
| Self-check | 1 | 1 | 1 |
| Calibration | 1 | | |
| Accuracy | 1 | | 1 |
| High frequency information | | 1 | 1 |
| Analog information | | 1 | 1 |
| Threshold signals | | 1 | 1 |
| Flexibility | 1 | | 1 |
| Reliability | 1 | | |

FIGURE F-1 (Cont)

Comparison of Candidate Systems

| | PCM/FM | PAM/FM/FM | HYBRID (PCM/FM/FM) (PAM/FM/FM) (FM/FM) |
|---|---|---|---|
| Maintainability | 1 | | |
| Logistics | 1 | | |
| Compatibility | 1 | | 1 |
| Power | 1 | | |
| Size and weight | 1 | | |
| Parallel data transmission | | | |
| Score | 15 | 6 | 10 |

(2) Data format. Because a major portion of the data will be in digital form and because of required accuracy of certain data (e.g., guidance parameters), a system which will accept and transmit digital data was chosen. The PCM system lends itself to various code formats which provide synchronization and reduction in total RF power requirements. The PCM format further provides compatibility with digital computers, which eases the data reduction burden.

(3) Bandwidth and Transmitter Power. To achieve an optimum system with respect to minimum launch-vehicle boost requirements, a minimum communications bandwidth must be maintained for the reason that an increase in bandwidth requires an increase in operating power, which in turn prescribes an increase in the weight of the orbital equipment.

A PCM/FM system will require approximately 1/8 the bandwidth of the FM/FM or the hybrid system. Through the use of orthogonal coding, PCM will require approximately 1/20 as much energy as a conventional FM/FM telemetry system in transmitting equal amounts of information. Discrete pulse position modulation requires high peak powers, and sub-carrier frequency modulation requires the generation of a large number of stable frequencies.

(4) Data Storage. It is assumed that continuous data transmission will be required during initial orbit, rendezous, and launch. During other periods it is projected that the data will be transmitted only at preset periods to preselected stations. To meet these requirements, two magnetic storage units with minimum time correlation capabilities will be required, one to handle digital data and the other to record analog signals. Because of high bit rate encountered during playback, a PCM system is superior.

(5) Self-Check Features. Because of the tie-in with the ground complex, the communications checkout equipment on board the OLF can be considerably reduced if the major portion of the communications checkout burden is placed with the ground. This philosophy should be incorporated into the OLF communications system. Failure detection is easier in a PCM system because of the go no-go characteristics of the digital circuitry.

(6) Calibration. Because of the required long life of the communication system, techniques which depend on calibration for data correlation are not suitable. A PCM system, because of its digital characteristics, most nearly meets the data correlation requirement. PCM exhibits none of the

errors due to zero drift, distortion, and sensitivity change seen in the FM/FM, PDM, and PAM systems.

(7) Accuracy. It is projected that the required data accuracy will fall into two ranges, one covering from 7 to 2 percent and the other from 1 to 0.1 percent. Through quantizing, either the PCM or hybrid system has this capability. The PCM system provides variable accuracies which can be programmed into the system through the control of the bit rates. The FM/FM system cannot provide the 0.1 percent accuracy required by many of the anticipated guidance parameters.

(8) High Frequency Information. The complexity in programming, multiplexing and encoding sets a practical limit on the range of information rates and accuracy that can be covered by PCM. Low accuracy, high frequency data is best transmitted by FM/FM or the hybrid system.

(9) Analog Information. The PCM system has no capabilities for transmission of analog information. No requirement for analog data transmission is anticipated for the OLF.

(10) Threshold Signals. A PCM system exhibits a well defined threshold which occurs when random noise exceeds one-half of the peak-to-peak signal input to the decision circuit for any significant amount of the detection time. The other systems can provide data of degraded quality in a high noise environment. Threshold signals are not expected, however, in operations involving the OLF where adequate transmitted power will be available to provide working safety margins.

(11) Reliability. Reliability is dependent upon the techniques used in the development and manufacture of a system and upon the type of system chosen. PCM logic lends itself readily to solid state circuitry which provides better reliability. In addition, marginal operation is less frequent in digital circuits. A more detailed analysis of reliability is given in Appendix G.

(12) Maintainability. By use of self-check features for detection and isolation of subassembly failures and by use of the OLF checkout equipment for failure detection and isolation of module failure, it is projected that a PCM system will require less maintenance than either of the systems which have the capability of handling analog information. Detailed maintenance and logistics concepts are given in Appendix H.

(13) Logistics. By the use of PCM digital building blocks identical to those used in other OLF equipment and by repair and maintenance at the subassembly and module level, it is felt that the spares requirements can be reduced substantially.

(14) Compatibility. Coded data will be used in most OLO operations; therefore, a PCM system is most compatible.

(15) Size and Weight. Microminiature circuit techniques can be employed to a greater extent in the PCM system; it is estimated that a 25 percent size and weight reduction can be achieved in the PCM system over the other two.

(16) Parallel Data Transmission. The PCM system has no capabilities for parallel data transmission; however, no such requirement is projected for the OLF telemeter.

FIGURE F-1 (Cont)

Comparison of Candidate Systems

| | PCM/FM | PAM/FM/FM | HYBRID (PCM/FM/FM) (PAM/FM/FM) (FM/FM) |
|---|---|---|---|
| Maintainability | 1 | | |
| Logistics | 1 | | |
| Compatibility | 1 | | 1 |
| Power | 1 | | |
| Size and weight | 1 | | |
| Parallel data transmission | | | |
| Score | 15 | 6 | 10 |

(2) Data format. Because a major portion of the data will be in digital form and because of required accuracy of certain data (e.g., guidance parameters), a system which will accept and transmit digital data was chosen. The PCM system lends itself to various code formats which provide synchronization and reduction in total RF power requirements. The PCM format further provides compatibility with digital computers, which eases the data reduction burden.

(3) Bandwidth and Transmitter Power. To achieve an optimum system with respect to minimum launch-vehicle boost requirements, a minimum communications bandwidth must be maintained for the reason that an increase in bandwidth requires an increase in operating power, which in turn prescribes an increase in the weight of the orbital equipment.

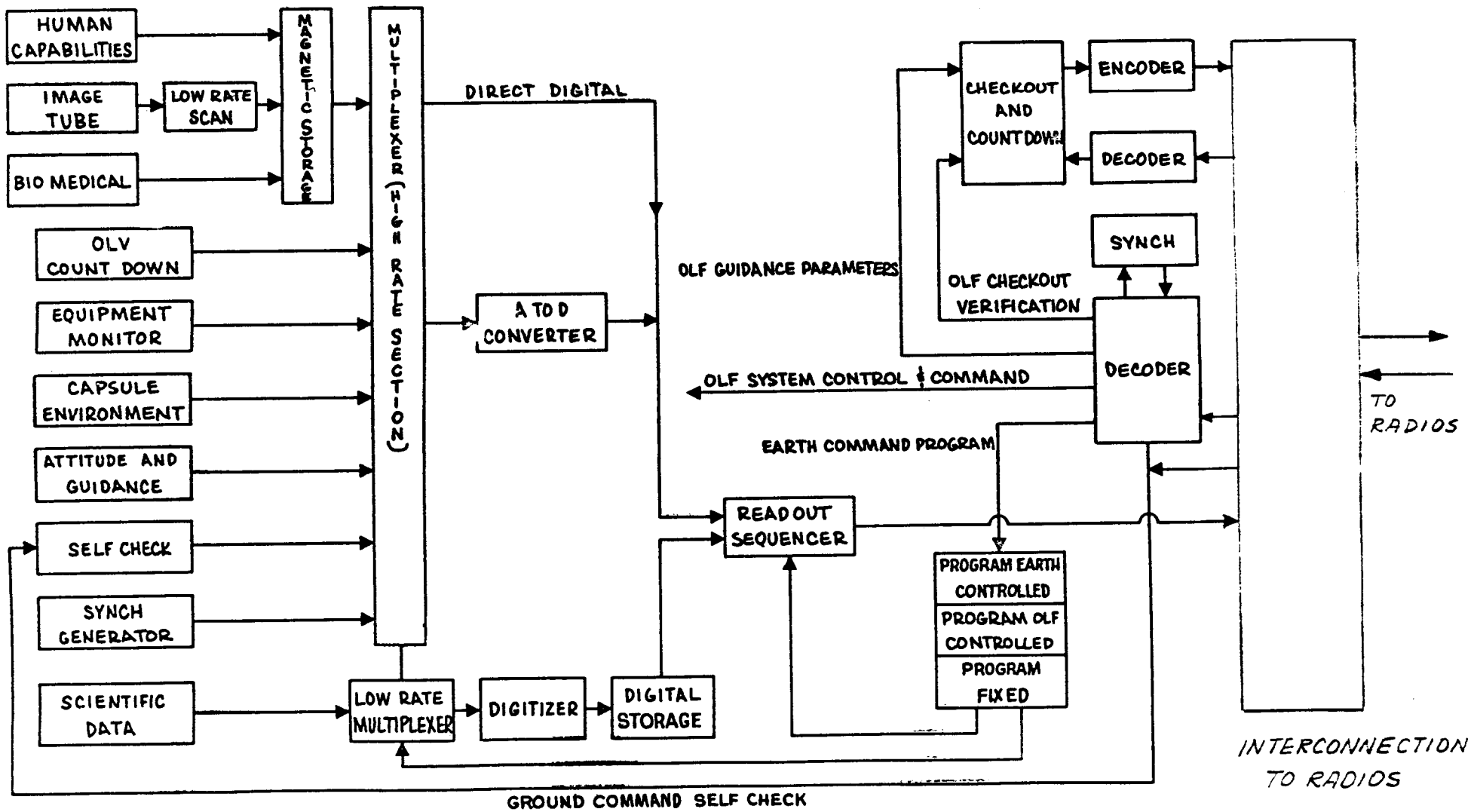



FIGURE F-2 PCM SYSTEM

FIGURE F-3

Projected Electrical Characteristics of
OLF Telemetry System

| Storage | Total Bits | Record Speed | Play Back Speed | | |
|---|---|---|---|---|---|
| Magnetic | 700 K | 200 B/Sec | 5 KB/Sec | | |
| Digital | 600 K | 15 B/Sec | 40 KB/Sec | | |
| A to D Converter | Signal Range | Rate | Max. Accuracy | Code Format | Bits per Word |
| | 100/1 | 40 KB/Sec | 0.1% | Binary | 5 to 10 |
| Multiplexer | Rate | Signal Range | Error | | |
| | 5 KC | 100 to 1 | 0.1% | | |

FIGURE F-4

Physical Characteristics - Basic Telemetry System

| Item | Assembly Number | Weight (Pounds) | Power (Watts) | Volume ($ft.^3$) |
|---|---|---|---|---|
| Signal Conditioning | 1 | 15 | 50 | 0.18 |
| Multiplexer | 4 | 60 | 28 | 0.72 |
| Sync Generator | 1 | 15 | 7.2 | 0.18 |
| Encoder | 2 | 30 | 14.4 | 0.36 |
| Programmer | 1 | 15 | 10.2 | 0.18 |
| Storage Unit (Electronics) | 1 | 15 | 22.2 | 0.18 |

FIGURE F-4 (Cont)

Physical Characteristics - Basic Telemetry System

| Item | Assembly Number | Weight (Pounds) | Power (Watts) | Volume ($ft.^3$) |
|---|---|---|---|---|
| Decoder | 1 | 15 | 14.4 | 0.18 |
| Checkout and Countdown | 1 | 15 | 20 | 0.18 |
| XMTR (Ground) | 0.5 | 7.5 | 5 | 0.09 |
| XMTR (S/C) | 0.5 | 7.5 | 22 | 0.09 |
| Receiver (Ground) | 0.5 | 7.5 | 20 | 0.09 |
| Receiver (S/C) | 0.5 | 7.5 | 10 | 0.09 |
| Power Supplies | 2 | 30 | 80 | 0.36 |
| Storage Unit (Mechanical) | | 3 | 4 | 0.10 |
| Antennas | | 20 | | |
| Cables and Connectors | | 30 | | |
| Totals (Basic Tele System) | 16.0 | 293.0 | 307.4 | 2.98 |

FIGURE F-5

Physical Characteristics
Additional Telemetry Equipment

| Item | Assembly Number | Weight (Pounds) | Power (Watts) | Volume ($ft.^3$) |
|---|---|---|---|---|
| Sensors | | 40 | 40 | 2 |
| Narro-band video (Transmit Only) | 2 | 30 | 50 | 0.36 |
| Control and display | | 20 | 20 | 2 |
| Totals | 2 | 90 | 110 | 4.36 |

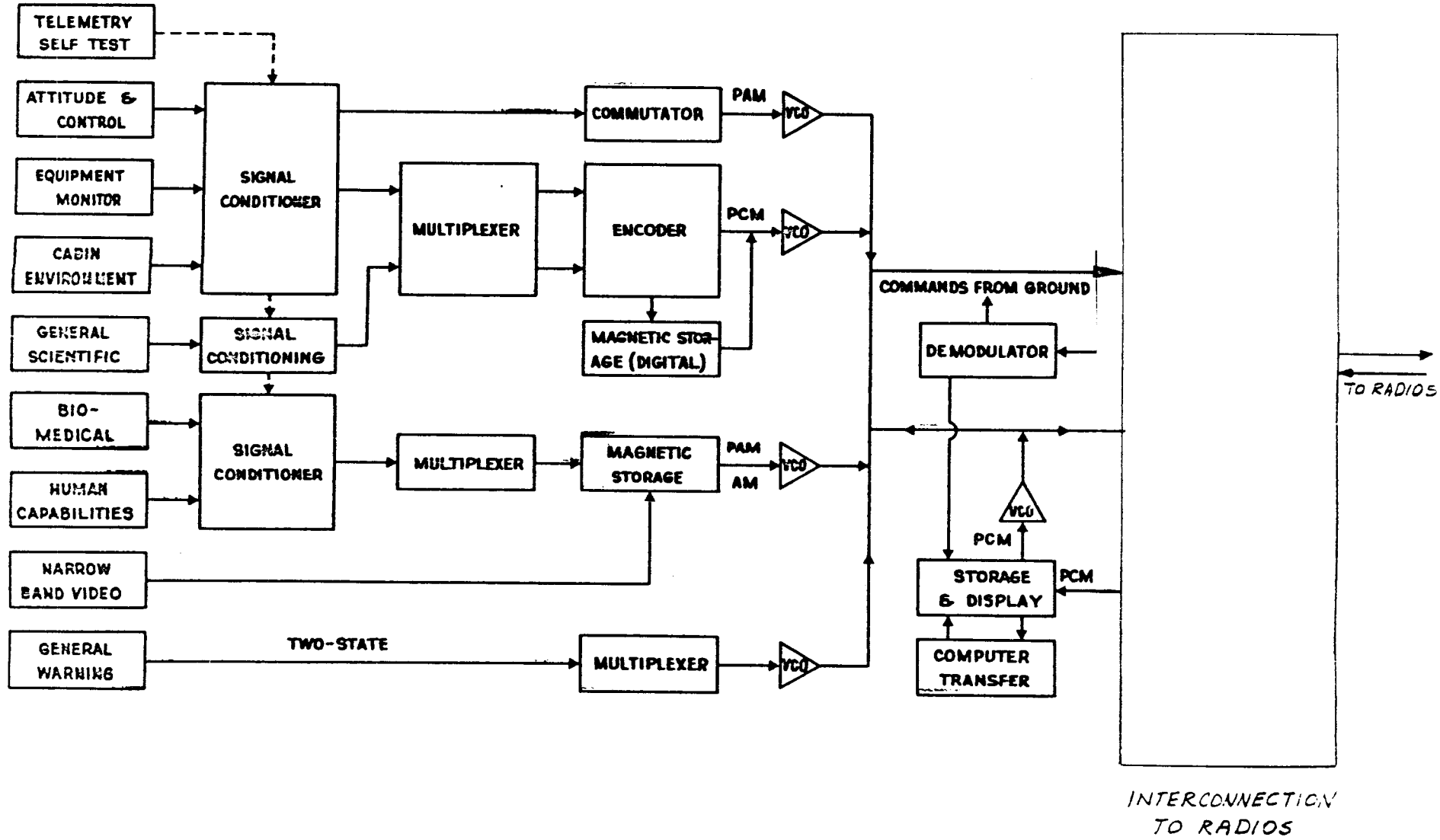



FIGURE F-6 ALTERNATE HYBRID SYSTEM

The estimated total values of weight, size, and power for the OLF communication system including spares are summarized below:

| | Basic System | Spares |
|---|---|---|
| Weight | 383 pounds | 219 pounds |
| Size | 7.34 ft.$^3$ | 3.99 ft.$^3$ |
| Power | 417 watts | 75 watts |

(1) Encoding. Straight binary coding will satisfy the OLF requirements, although the wide range of signal accuracies, signal levels, and signal information rates coupled with identification difficulties might make the use of redundant coding schemes mandatory. The binary format to be used will require a 14 bit word. Ten bits will be required for data identification, three bits for word identification, and one bit for parity. Coding of binary data to an accuracy of 0.1 percent requires a minimum 10 bit data word. The 10 bits used to identify the data signal amplitudes provide readout to 1/1024 part, which provides an accuracy of better than 0.1 percent for the high level signals. The threshold of the comparator amplifier in the encoder (see figure F-7) will limit the accuracy obtainable from the low-level, 0 to 50 mv signal. This threshold limitation makes only the first eight bits significant or limits identification of the data signal to 1/256 part. This provides an encoding accuracy of 0.4 percent for the low level signals.

The output of the multiplexer will be introduced into the encoder assembly (see figure F-7). The sample and hold circuit will look at this signal and store the voltage

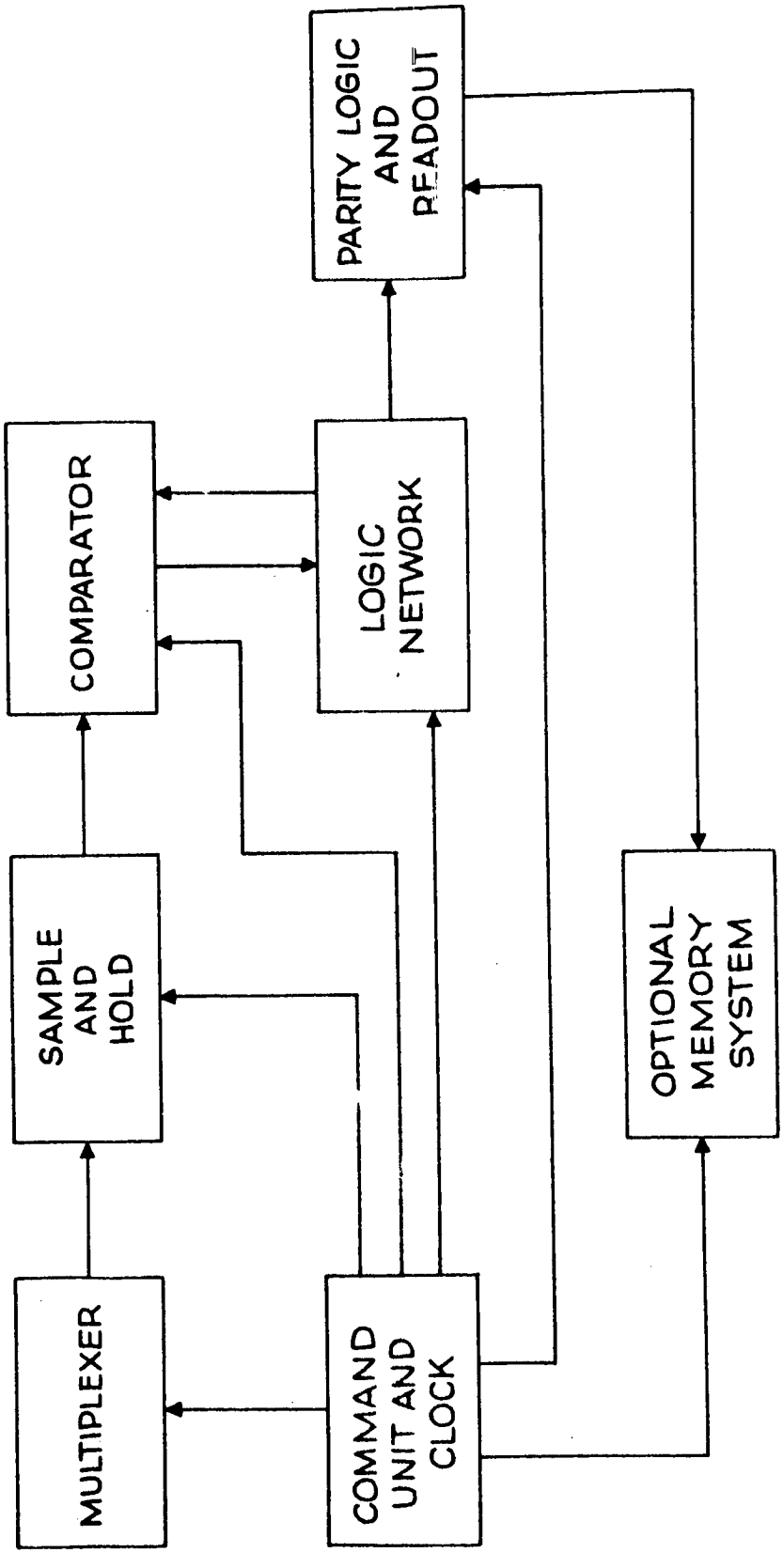



FIGURE F-7 DECODER SYSTEM

on a capacitor through a compensated emitter follower amplifier. At the end of the sampling period, the charge stored on the capacitor will set the level of the output on the hold amplifier, which will be fed into one side of the comparator amplifier. A reset pulse will activate the digitizer circuit in the logic network, the output of which will provide a reference input to the comparator amplifier. The high and low-level switches will be synchronized through the programmer to allow the comparator amplifier to operate either on a low level 50-mv full scale voltage or a high level 5-volt full scale voltage. This automatic level and accuracy ranging will eliminate the need for a precision dc voltage amplifier ahead of the analog-to-digital converter and will aid in minimizing over-all system errors. The comparator will have a regenerative type output circuit that decides which of two inputs is larger. If the input from the sample and hold is larger than the input from the digitizer circuit, the comparator will not provide an output pulse. The next most significant bit (1/4 full scale voltage) is then added to the most significant voltage. Should the input from the digitizer network now be larger than the input from the sample and hold, the comparator will produce a pulse output to the encoder to remove the 1/4 full scale voltage from the comparator. This process of comparing continues through the full scale of ranges: 1/8, 1/16, 1/32, and so on. Each time the comparator generates a pulse, the voltage introduced from the digitizer circuit discrete scale is removed from the comparator, and when the comparator does not generate a pulse, the discrete scale voltage is added to the input to the comparator. This is repeated for the 10 bit encoding of the analog sample, and a parity bit is added in the serial pulse train from the parity logic circuit. Frame synchronization codes will be generated

by a frame sync logic circuit. The frame sync will be started by a reset output from the commutators and will provide one word for frame sync after each of the data channels has been sampled.

(2) Multiplexing. It is expected that the multiplexing unit will be extremely flexible to fulfill system requirements, including those concerning a wide range of input signals, sampling rates, and operating modes. All solid-state devices will be used, and both super- and subcommutation techniques together with different rate sections will be employed.

The multiplexer will be built in sections. Each section will be programmed to operate on a predetermined sampling plan. This sampling plan may be changed on a time-base command or on an earth-to-vehicle command. The predetermined sampling plans will include a discrete number of individual channel sampling rates in any frame to make the sampling rate compatible with the frequency content of the sample. The sampling plan will take into account data priorities and available transmission bandwidth. The multiplexer will have a low level section (0 to 50 mv) and a high level section (9 to 5 V). Figure F-8 presents a block diagram of the multiplexer and the auxillary equipment associated with it.

(3) Electrical Characteristics. The multiplexer will use a basic solid-state switch consisting of two silicon surface alloy junction transistors with the collectors common. This permits the two emitters to be used as the input and output terminals. See figure F-9 for the basic switch configuration. $E_c$ will be a square wave switching voltage supplied from a programmed matrix. When the switch is open, the polarity of the switching voltage E is as shown, and all

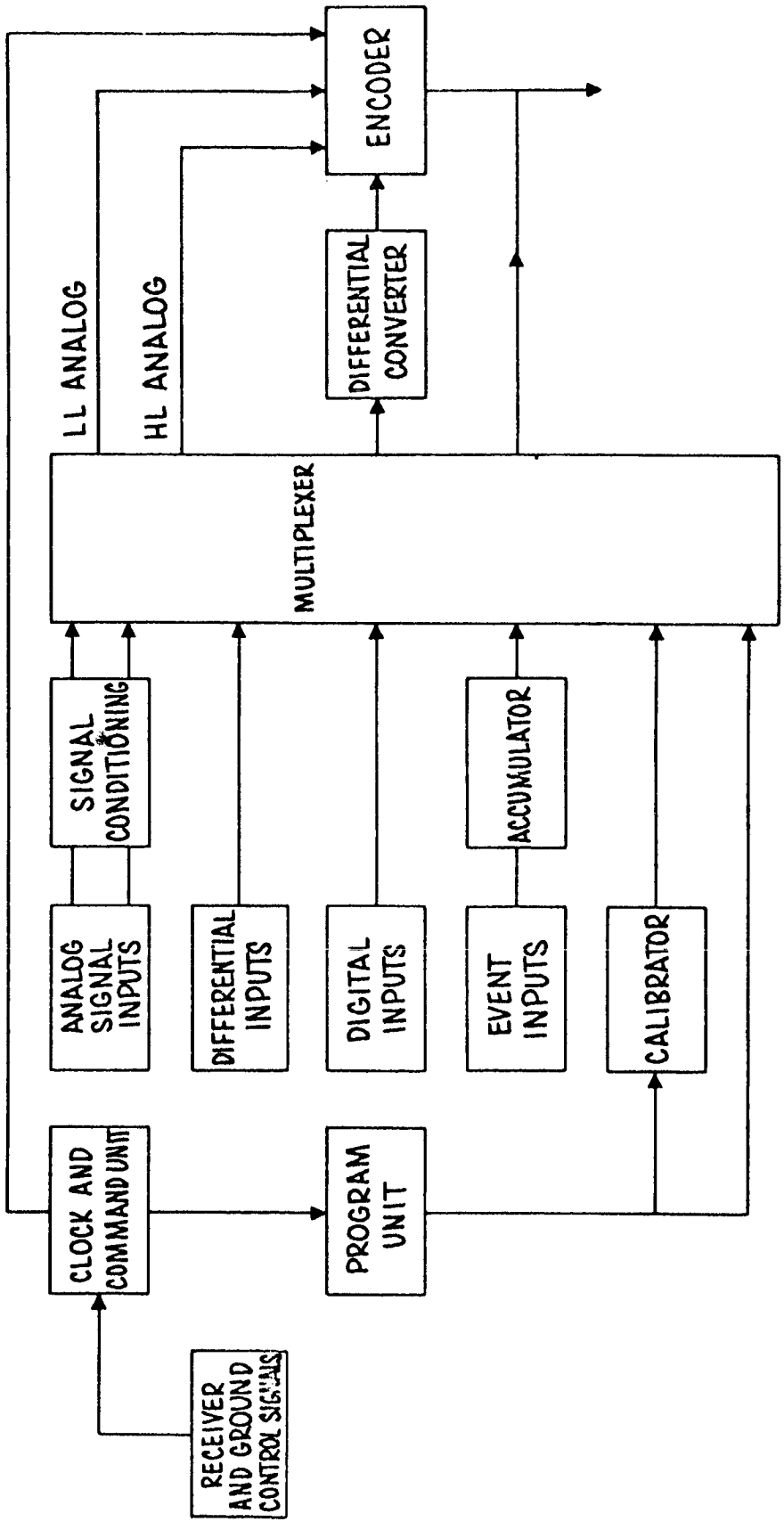



FIGURE F-8 PROGRAMABLE MUTIPLEXER SECTION

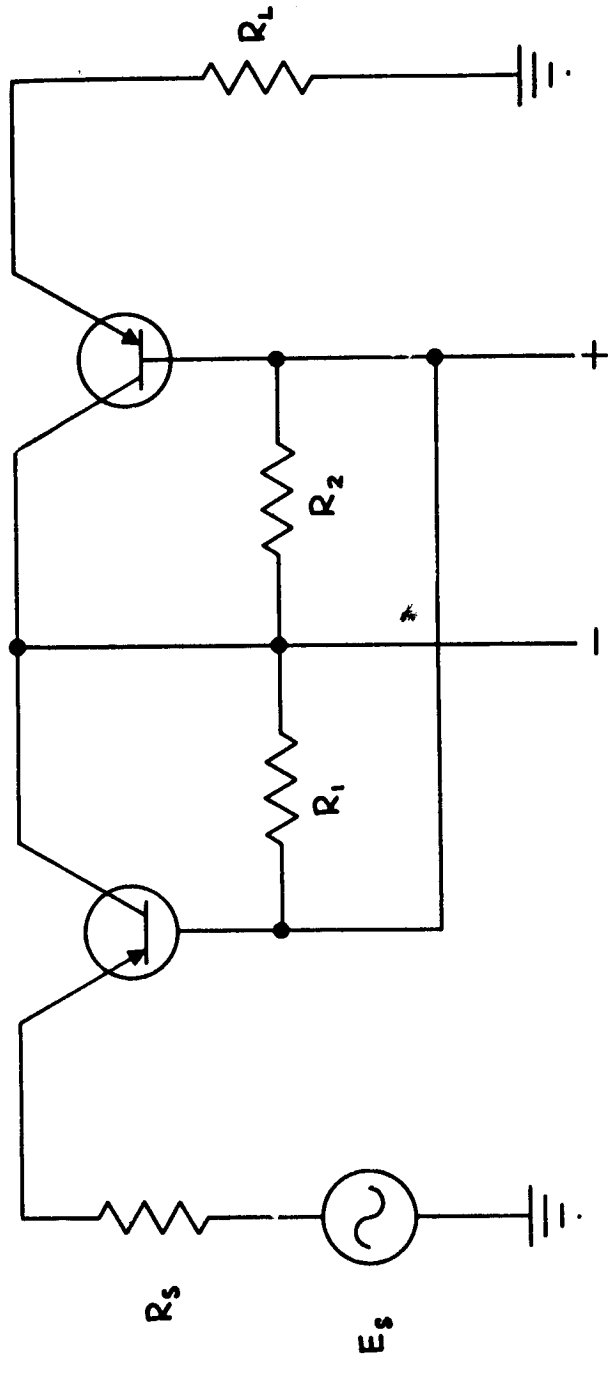



FIGURE F-9 BASIC SWITCH

junctions are reverse biased. The leakage of a back-biased collector junction is 20 times that of the emitter junction. Therefore, using the transistor with the emitter junction in series with the signal and load results in a resistance more than 20 times that of the collector junction. The magnitude of this resistance is greater than 10,000 megohms. When the switch is closed, a low impedance path will exist between the source voltage and load resistor through the saturated junctions of the two transistors. Matched transistors with minimum saturation voltages will be selected. The inverse connection will also result in a greater isolation during switch off time. The switch must also maintain voltage balance over large variations in temperature. This balance can be maintained since both transistors are made of the same material and the junctions will be located in close proximity to each other. The saturation voltage on the transistors will be nearly equal, and the difference will be in tenths of millivolts. Consequently, for a large change in temperature the difference signal and the error signal introduced will be on the order of microvolts.

Sequencing will be provided by a programmed multiplexer driver (see figure F-10). The driver is controlled by a series of gates and a control pulse. Figure F-11 shows the pulse waveform for synchronizing and driving. Complete dc isolation must exist between the sequencing circuitry and the signal path to eliminate the chance of cross coupling. This isolation will be provided through the use of transformer coupling.

(4) Signal Conditioning. For this system a minimum amount of signal conditioning equipment is anticipated. The majority of operations in this area will be performed by

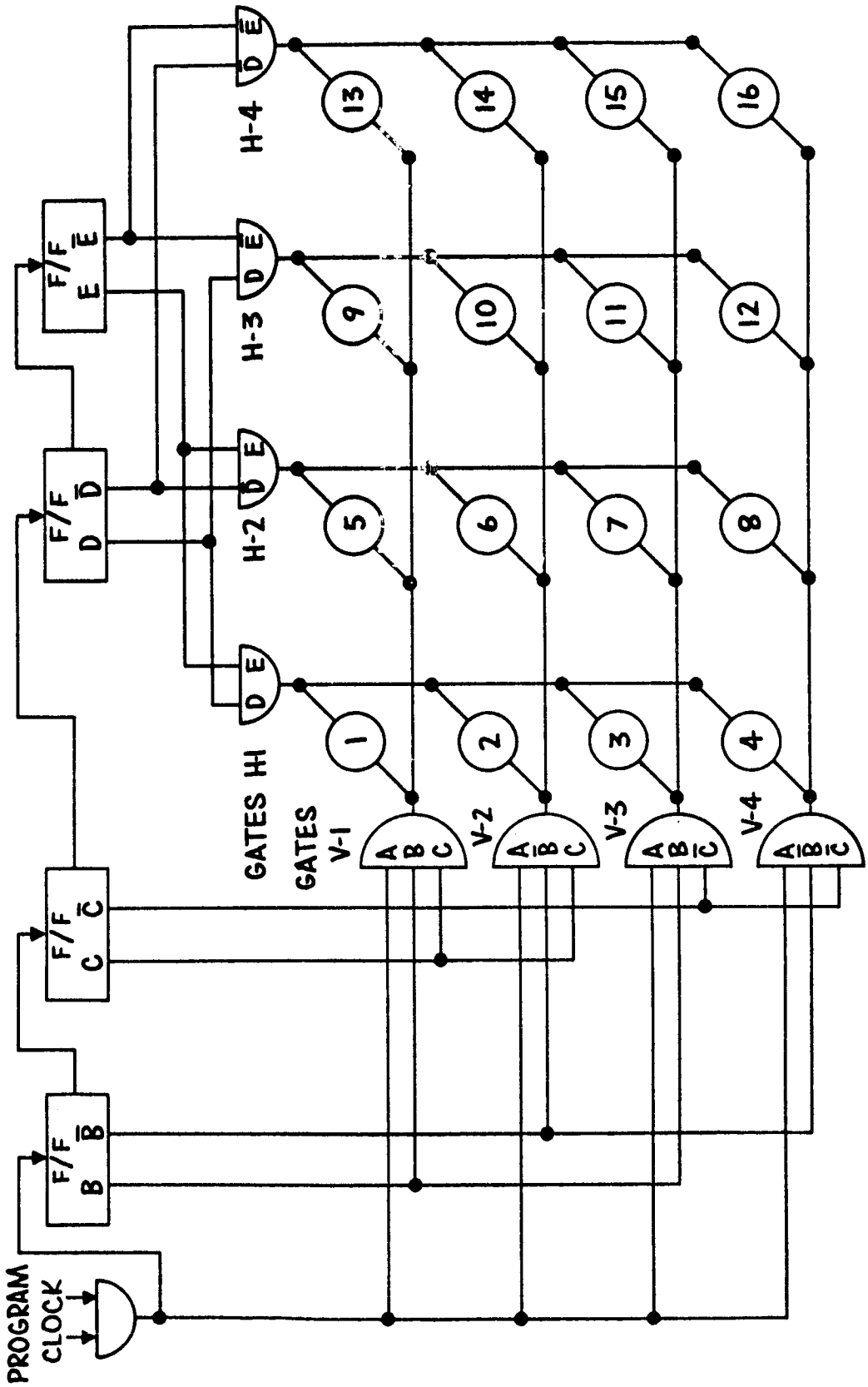



FIGURE F-10 PROGRAMMED MULTIPLEXER DRIVER

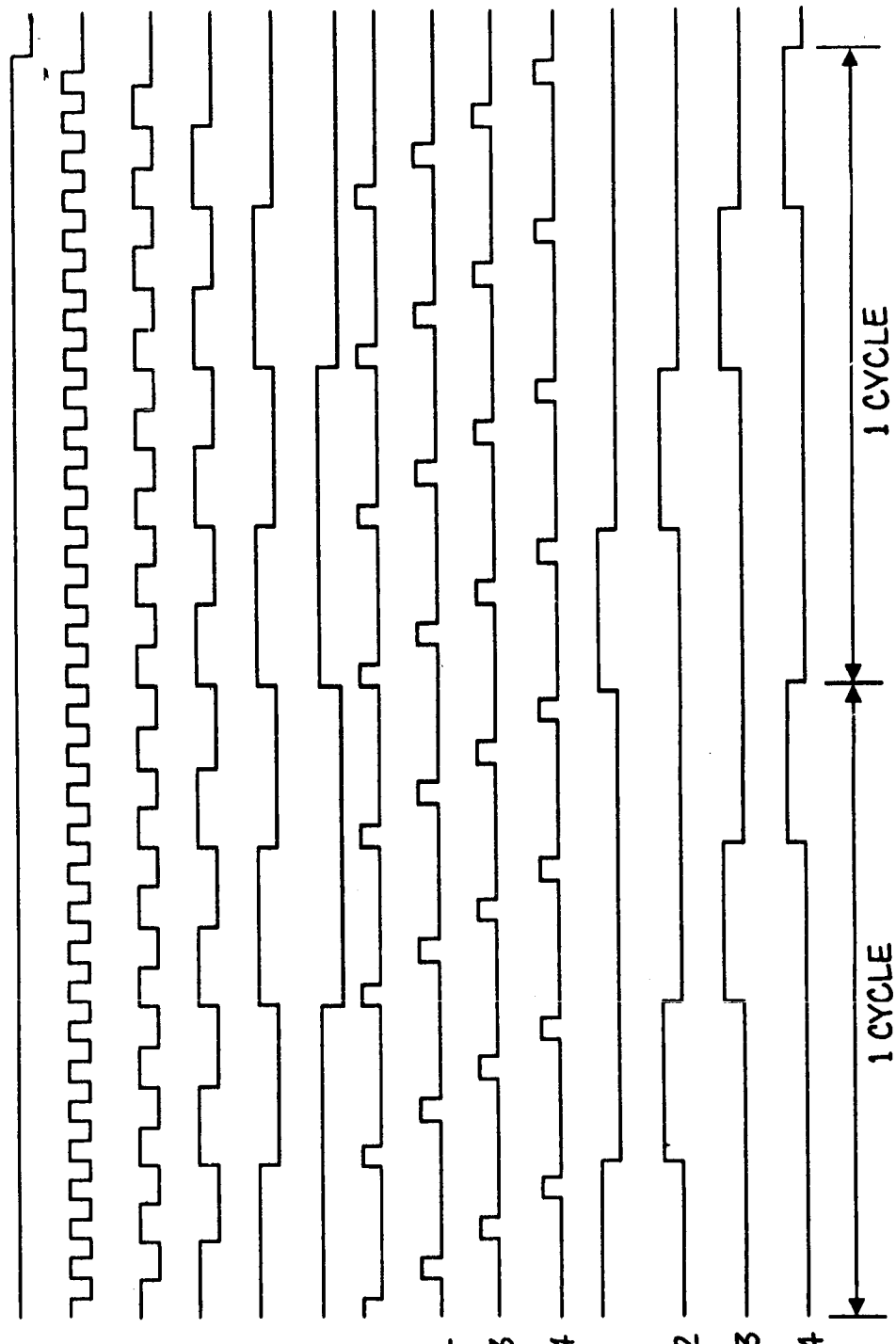



FIGURE F-11 WAVEFORMS OF DRIVER UNIT

automatic level and accuracy ranging in the encoder. Signal conditioning may include phase demodulators, resistive divider networks, magamps, matching networks, and that conditioning necessary to make any special sensor compatible with the telemetry measurement schedule.

Measurements with a differential output such as bridge networks will be processed through a differential converter. The converter will have a constant gain and will provide a high common mode rejection. The input to the normally ac coupled transmitter requires special conditioning to accept a random pulse train of a non-return-to-zero format. Conditioning is necessary to regenerate pulse train signals having varying average dc levels into pulse train signals with a constant dc reference level. The transmitter will then see the "go" or "no go" condition of the last bit, and no dc shift will be experienced by the transmitter when a nonsymmetrical pulse train is applied. This go no-go feature results in polarity synchronization of the transmitter with respect to the input signal from the encoder in the event of an interrupted pulse train or loss of power.

(5) Storage. The OLF will provide storage for four types of data: human capabilities and bio-medical information, picture images, scientific data and checkout and countdown information. It is anticipated that only periodical monitoring of human capabilities and bio-medical information will be required after Phase I in view of knowledge gained during Phase I and previous programs. The results from these periodical tests will be recorded on a controlled, continuous record, magnetic tape. The operator controlling the test will likewise control the record time and rate. The picture image data will also be recorded on this tape.

Scientific data, for the most part, will change slowly with respect to time. This data will be converted to binary format and stored on a magnetic tape with a stepping-type transport which advances upon command. Playback for both recorders will be programmed for relaying stored information to preselected ground stations. Maximum storage time will be 24 hours. Checkout and countdown information will be transferred directly into and from the OLF checkout computer.

(6) Self-check Features. Since a malfunction in the OLF telemetering equipment could generate extraneous signals which have the same characteristics as measured information, some means must be provided to allow the ground equipment to distinguish failures. By periodically transmitting a set of properly selected self-check signals, a ground station can detect a malfunction and isolate the fault to the subassembly level. Wherever possible, this feature will be incorporated into the OLF communications system to avoid excessive weight of checkout equipment in the OLF.

(7) Power Supply Requirements. Power for the telemetry system will be supplied through a dc to dc converter with a multiwinding output. Each output winding will supply the necessary voltage level and isolation for a given subsystem. Balanced diode quads will be used in rectification to ensure minimum offset flux being produced by unbalanced loading. Voltage regulator sections will be provided, where needed, for precision reference sources. Power supply filtering will be used to minimize RF interference (conducted or radiated) being fed to the primary power source.

# APPENDIX G

# RELIABILITY DEVELOPMENT PROGRAM

## G-1. INTRODUCTION

To ensure mission success through equipment reliability, it is necessary to define and pursue a program which will optimize the overall reliability of the system.

## G-2. OBJECTIVES

In order to adequately define the proposed reliability program, the following objectives were established and met. The reliability requirements for the communications equipment for each phase of the OLO program were determined. Problem areas associated with the reliability effort were defined, and solutions to these problems were proposed. Then the reliability program deemed necessary to achieve the reliability goals for each phase of the OLO program was determined.

## G-3. APPROACH

### a. General

By necessity, reliability is of prime importance for the OLO mission from considerations of mission success, human safety, and program cost. To achieve maximum overall

system reliability, both the inherent reliability and the operational reliability of the equipment must be optimized. Inherent reliability can be defined as the potential reliability associated with the basic equipment design. Design techniques that can enhance reliability of the equipment are circuit and/or component redundance, component derating, fail-safe circuitry, and self-healing circuitry. Operational reliability, which can be considered as an extension of the inherent reliability, can be defined as the actual or available reliability of the equipment after fabrication. Thus, the operational reliability is always less than the inherent reliability. Some of the factors contributing to this decrease in reliability experienced in the field are inadequate fabrication processes, errors in human judgement, and unpredictable equipment degradation with time.

To ensure mission success through equipment reliability, a program will be pursued which will optimize the overall system reliability. This will be effected by utilizing fabrication and design techniques that maximize the inherent reliability of the equipment and minimize the effects of inadequate fabrication processes, errors in human judgement, and the degradation effects associated with time.

Initially, the POMS for each phase of the OLO program will be determined. This probability will be a function of environmental stress levels, equipment operating time, and various equipment redundancies and will vary as man's role changes in the maintenance and logistics pattern. The operating reliability will then be the optimum compromise of each of these areas for each phase of the program.

b. Reliability Analysis

Basic to the discussion of reliability is that the total reliability, $R_T$, of any configuration is the products of the individual reliabilities, $R_i$, of its constituents; that is,

$$R_T = R_1 \cdot R_2 \cdots R_n = \prod_{i=1}^{n} R_i \qquad (1)$$

Implicit in equation (1) is the requirement that the basic elements $R_i$ be independent in the statistical sense. Except in special cases where interaction can be shown to exist between the $R_i$'s, independence is a defensible and commonly made assumption.

Simple systems with few elements have the possibility of much higher inherent reliabilities than highly complex systems with many parts. Unfortunately the requirements placed upon the systems under proposal demand that the systems be of severe complexity, resulting in the ultimate need for high component reliabilities.

For $R_T$ to be maximized, it is not only necessary that all of the $R_i$'s be large, but it is equally important that they have approximately the same value. For example, efforts to raise the total reliability by increasing some of the contributing reliabilities, for example from 0.99 to 0.995, are ineffectual when there are remaining reliabilities as low as 0.7 or 0.8 in the system. Although high, homogeneous reliabilities are desired, they are not generally realizeable simply because the contributing items are designed for such varied functions. To raise the reliability of an item above a given level is most costly and in some cases, with present or even proposed techniques, extremely difficult.

To overcome this problem it is common to have replacement parts for those whose reliability is low. This philosophy of redundance admits of two extremes:

Case I - The total configuration, $R_T$, considered and packaged as a single block, has a single spare $R_T$.

Case II - The total configuration, $R_T$, considered and packaged as n blocks, has a single spare for each of the n blocks. In each case the total spares are equal to $R_T$. From reliability considerations alone, what degree of redundancy should be

$R_T$ $\qquad\qquad$ $R_1$ $\quad R_2 \ldots\ldots\ldots R_n$ $\quad R_T$

$R_T$ $\quad$ Spare $\quad$ Spares $\quad$ $R_1$ $\quad R_2 \ldots\ldots\ldots R_n$ $\quad R_T$

Case I $\qquad\qquad$ Case II

adopted with the multiplicity of combination for grouping that exists?

To answer this question it is necessary to restate the problem in mathematical terms. A basic block of equipment has a reliability P. It is desired to achieve a higher reliability R by dividing the block into N arbitrary sections and by having replacements for n of the modules which occupy these sections. What is the optimum number of sections, N, and replacements, n, required to achieve this goal?

1 2 3 .. $x_i$ .. n n+1 ..... N Basic block P

1 2 3 .. $x_i$ .. n Spares R

Let P = reliability of basic block

N = total number of sections

n = number of modules which have replacements

N-n = number of modules without replacements

$x_i$ = reliability of module in ith section

$P_R$ = overall reliability of redundant sections

$P_N$ = overall reliability of non redundant sections

R = reliability revised block with redundance.

Since it is assumed that a failure in any one of the modules in the basic block will cause the equipment to fail, the following equation holds

$$P = \prod_{i=1}^{N} x_i \qquad (2)$$

Now consider the ith section; assume it has a spare module. The module for this section and its replacement both have a reliability $x_i$. For this section to fail it is necessary that both modules fail. The probability of this event is given by $(1 - x_i)^2$. Therefore, the probability of success of this section is $1 - (1 - x_i)^2$. This holds for each section which has a spare. Therefore,

$$P_R = \prod_{i=1}^{n} \left[1 - (1 - x_i)^2\right] \tag{3}$$

The reliability of the non redundant sections is given by

$$P_N = \prod_{i=n+1}^{N} x_i = \prod_{i=1}^{N-n} x_i \tag{4}$$

The overall reliability of the revised block is then

$$R = (P_N)\,(P_R) = \prod_{i=1}^{N-n} x_i \prod_{i=1}^{n} \left[1 - (1-x_i)^2\right] \tag{5}$$

In order to make calculations possible and to get an insight into the problem, assume $x_i = x_j = x$ for all i, j. Then

$$R = x^{N-n} \left[1 - (1 - x)^2\right]^N \tag{6}$$

where

$$x = P^{1/N} \tag{7}$$

c. Results

Figures G-1a, G-1b, G-2a, and G-2b illustrate the achieved increase in reliability as a function of N for basic reliabilities of 0.5, 0.9, 0.95, and 0.99, respectively. Each figure has curves for 25, 50, 75, and 100 percent redundancy. These curves show immediately that reliability increases rapidly with increasing redundancy and rather slowly with increasing N. For example, with a basic reliability of 0.9, division into 4 sections and 100 percent redundancy increases the

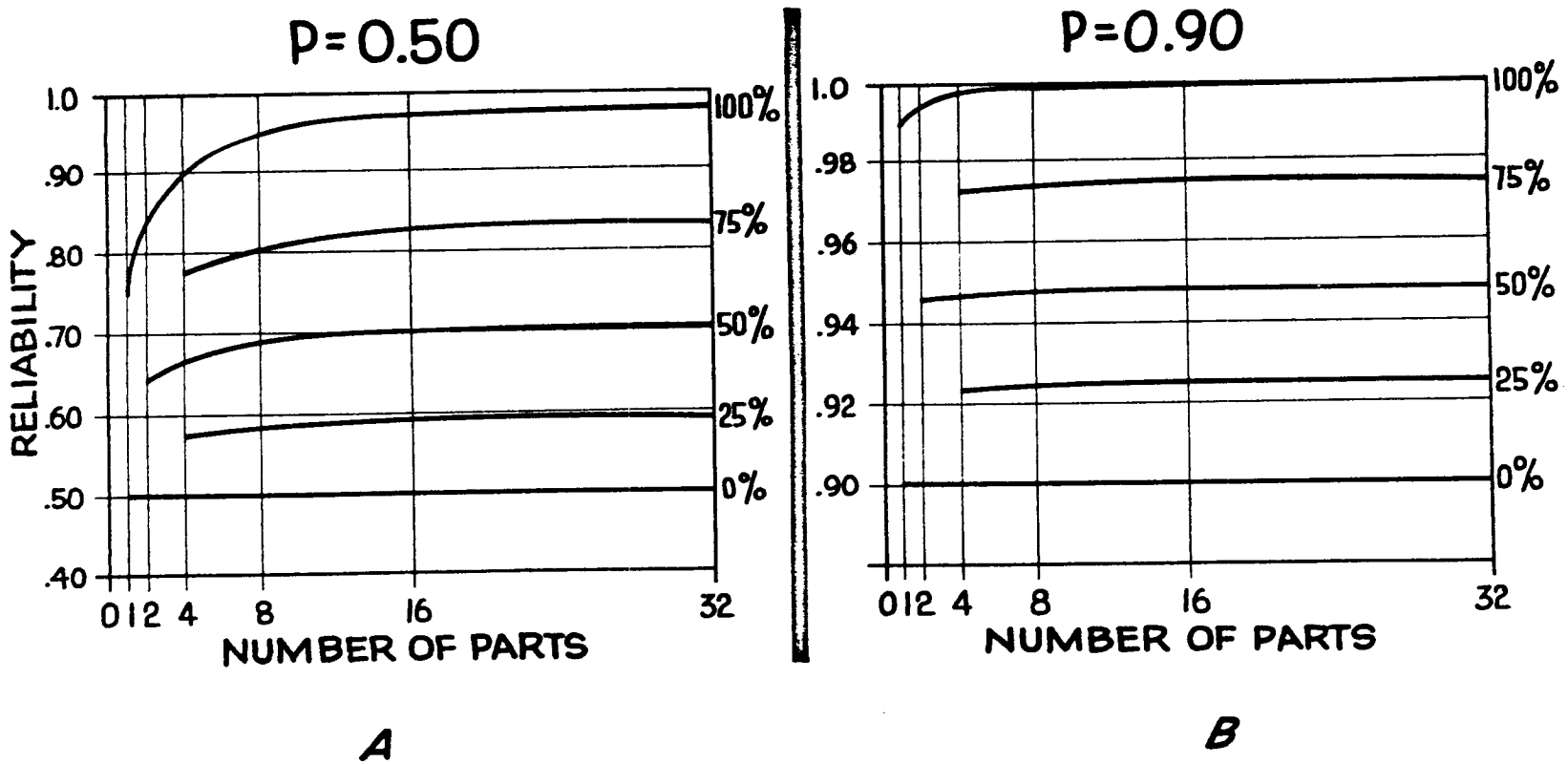



FIGURE G1 BASIC RELIABILITY P=0.50 AND P=0.90

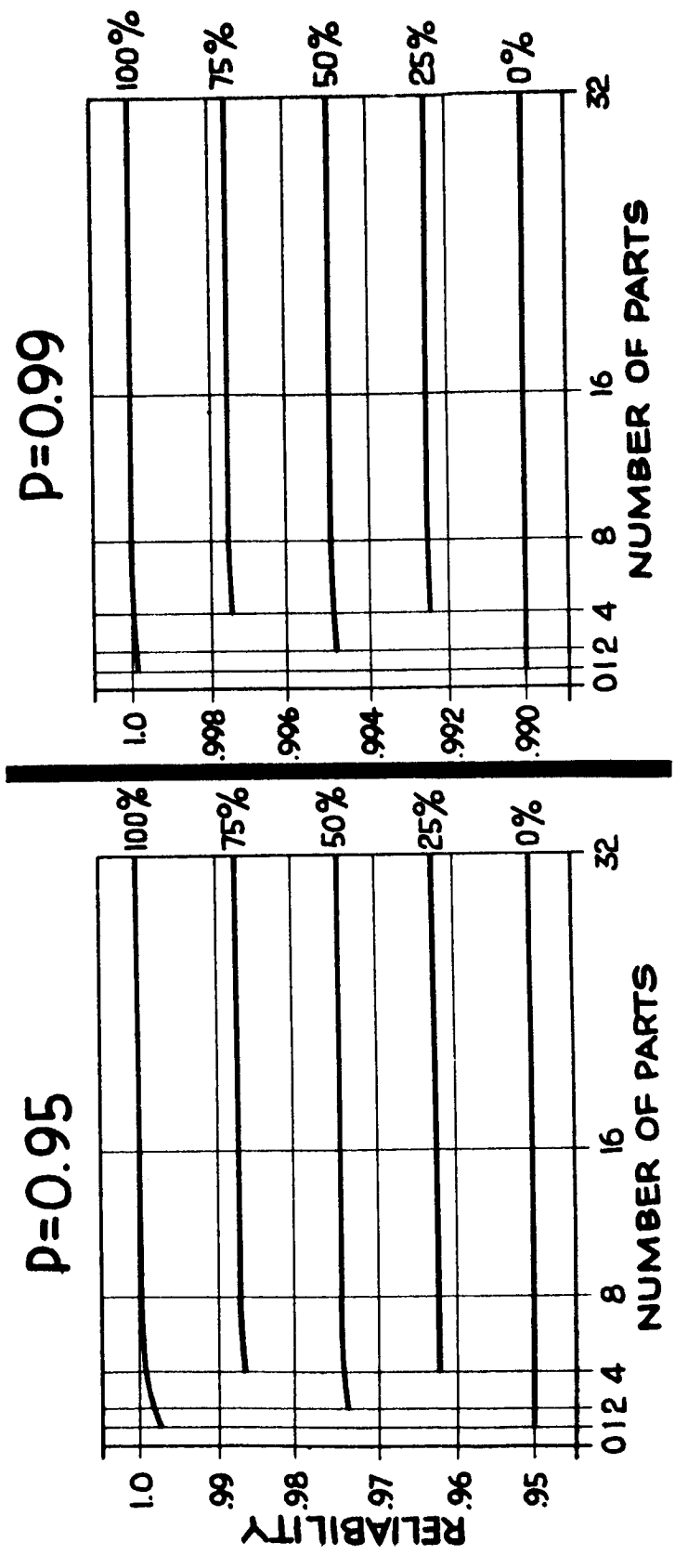



FIGURE G-2 BASIC RELIABILITY
P=0.95 AND P=0.99

reliability of 0.997, whereas division into 32 sections with 50-percent redundancy increases the reliability to only 0.948.

Figure G-3 shows the variation of reliability with weight for prescribed initial reliabilities and percent of redundancy. It is important to observe that 100-percent redundancy incorporated into the design renders the initial reliability of the configuration relatively insensitive to the final reliability. Because the percent of redundancy is such a critical parameter and the weight w for the telemetry system is comparatively small, at least 100-percent redundancy is proposed to achieve the desired reliabilities.

Consider a block of equipment with a basic reliability P. Let this block consist of N identical modules. Let the number of spares be n. Since the spares are completely interchangeable, the new reliability is given by

$$R = 1 - \sum_{s=n+1}^{N} \binom{N}{s} p^{N-s} q^{s} \qquad p = p^{1/N} \tag{8}$$

where p is the reliability of each module
and q is the probability of failure for each module (1 - p)

In contrast to this let another block consist of N different modules, each of which has the same reliability p. Then the reliability becomes

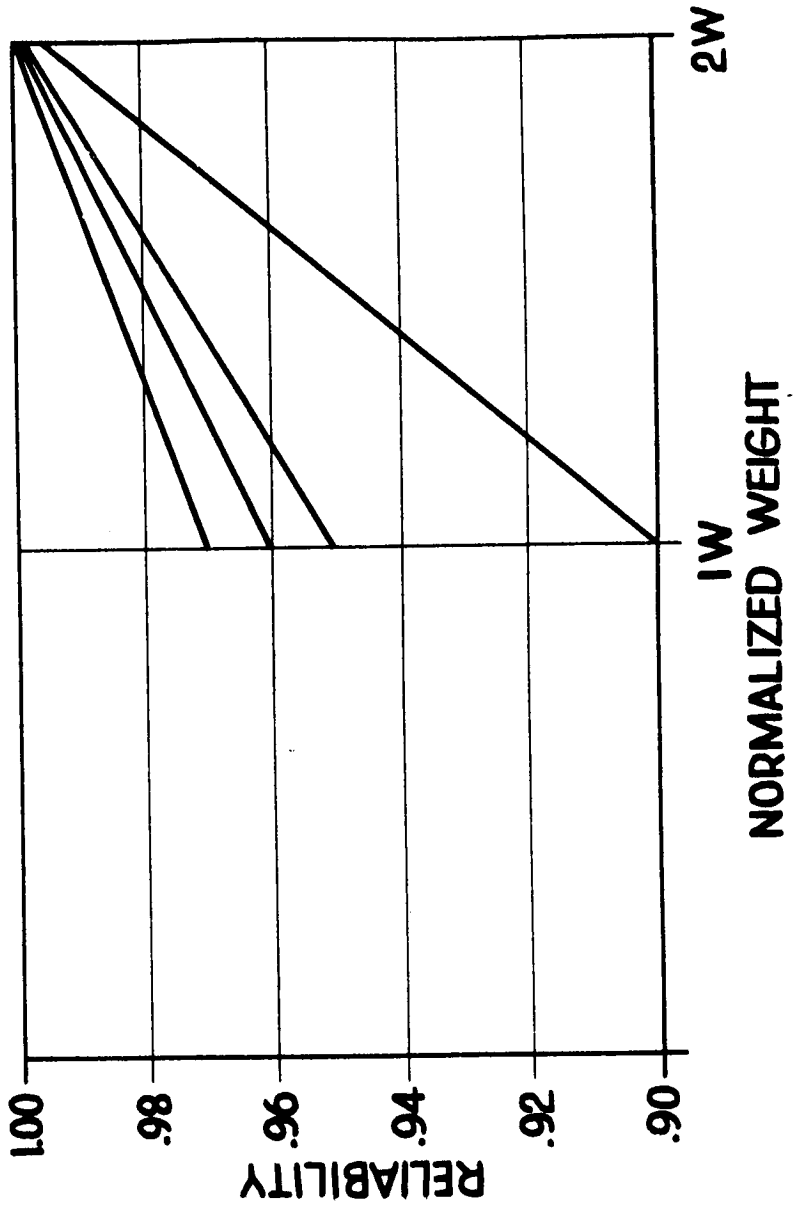



FIGURE G-3 RELIABILITY

$$R = p^{N-n} \left[1 - (q^2)\right]^n$$

where p and q are defined above and p is still given by

$$p = p^{1/N}$$

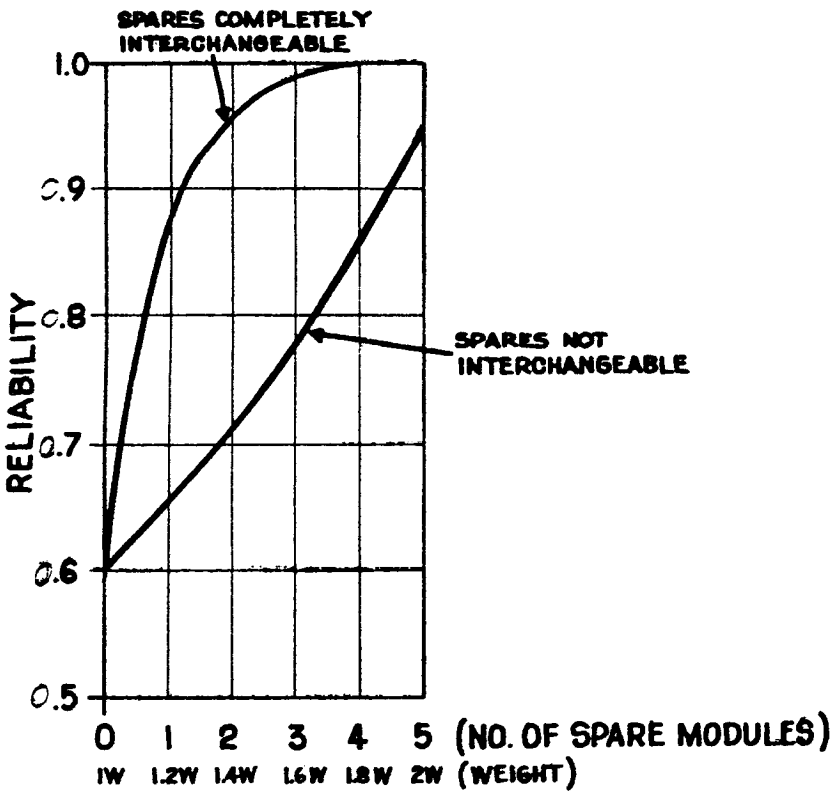
The advantages of using identical modules wherever possible are shown very dramatically in figure G-4a and G-4b. For a block of five identical modules with a basic reliability of 0.9, a redundancy of 20 percent gives a reliability of 0.994, and a redundancy of 40 percent yields a reliability of 0.9997. Even when the basic reliability is 0.59, less than 100-percent redundancy brings the final reliability very close to unity. It is thus apparent that the use of identical circuits whenever possible allows a minimum number of spares and thus minimizes weight and space.

Consider now the case of multiple redundancy with noninterchangeable items. The reliability of a block consisting of N items is given by

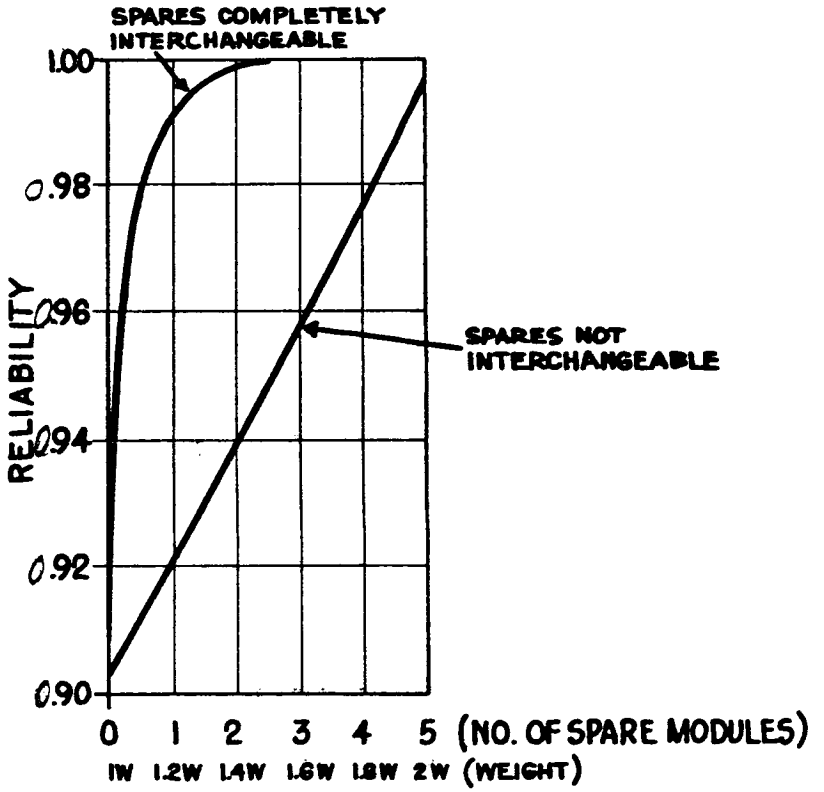
$$R = \prod_{i=1}^{N} P_i \tag{9}$$

and the weight of the block by

$$W = \sum_{i=1}^{N} w_i \tag{10}$$




A




B

FIGURE G-4 RELIABILITY-SPARES INTERCHANGEABILITY

where $p_i$ is the reliability of the ith item and $w_i$ the weight of the ith item. If there is a spare for n of the items, then the total reliability is

$$R_T = \prod_{i=1}^{N-n} P_i \prod_{j=1}^{n} \left[1 - (1-P_j)^2\right] \qquad (11)$$

and the weight is

$$W_T = \sum_{i=1}^{N-n} w_i \sum_{j=1}^{n} 2w_j \qquad (12)$$

where $p_i$ is the reliability of the ith item which has no spare, $p_j$ is the reliability of the jth item which has a spare, and $w_i$ and $w_j$ are the corresponding weights.

If each of the n redundant items has L spares, then equation (11) would become

$$R_T = \prod_{i=1}^{N-n} P_i \prod_{j=1}^{n} \left[1 - (1 - P_j)\, L + 1\right] \qquad (13)$$

and

$$W_T = \sum_{i=1}^{N-n} w_i + \sum_{j=1}^{n} (L + 1)\, w_j \qquad (14)$$

Since $p_j < 1$, it is obvious that as L increases $(1 - P_j)^{L+1} \rightarrow 0$ and equation (13) would become, in the limit,

$$R_T = \prod_{i=1}^{N-n} p_i \qquad (15)$$

but (14) remains

$$W_T = \sum_{i=1}^{N-n} w_i + \sum_{j=1}^{n} (L+1)\, w_j \qquad (16)$$

Although $R_T$ approaches a limit as L increases, $W_T$ does not but rather increases without bound.

For convenience, consider the case where each item has the same reliability and the same weight:

$$P_i = P_j = P \quad i = 1, \ldots, N-n \quad j = 1, \ldots, n$$

and

$$w_i = w_j = w \quad i = 1, \ldots, N-n \quad j = 1, \ldots, n$$

then equation (13) is

$$R_T = p^{N-n} \left[1 - (1 - p)^{L+1}\right]^n \qquad (17)$$

and equation (14) is

$$W_T = (N-n)\, w + n\, (L + 1)\, w$$

$$W_T = (N + nL)\, w \qquad (18)$$

Let p = 0.974 be the reliability and $W_o$ the weight of each of four items in a block. Then without redundancy

$$R_T = P^4 = 0.9$$

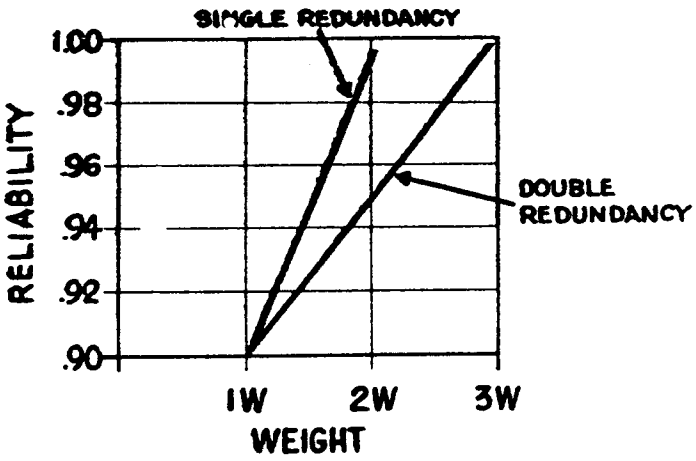
$$W_T = 4W_o$$

The increase in $R_T$ and $W_T$ for 0, 25, 50, 75, and 100-percent single and double redundancy is shown in figure G-5 and figures G-6a and G-6b.

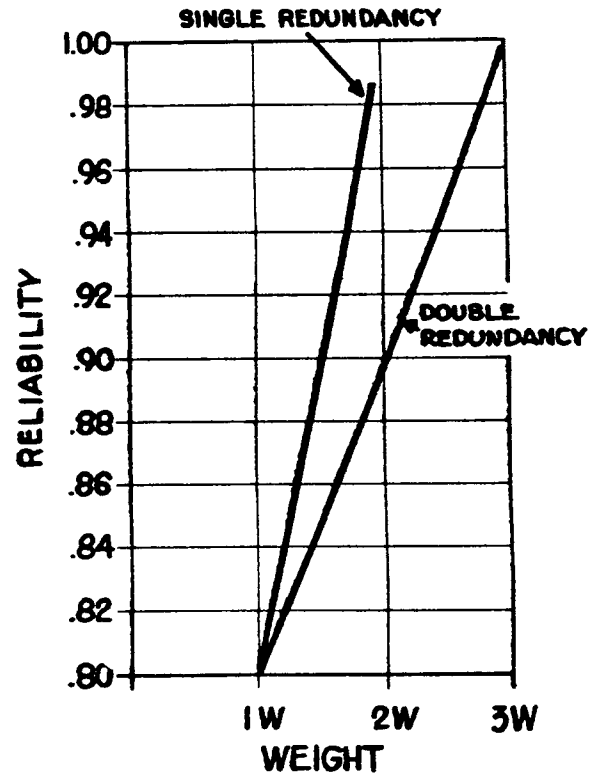
Figure G-5

| Single Redundancy | | | Double Redundancy | | |
|---|---|---|---|---|---|
| % Redundance | $R_T$ | $W_T/W_o$ | % Redundance | $R_T$ | $W_T/W_o$ |
| 0 | 0.9 | 4 | 0 | 0.9 | 4 |
| 25 | 0.923 | 5 | 25 | 0.924 | 6 |
| 50 | 0.947 | 6 | 50 | 0.949 | 8 |
| 75 | 0.972 | 7 | 75 | 0.974 | 10 |
| 100 | 0.997 | 8 | 100 | 0.999 | 12 |

Figure G-5 shows that the greatest reliability is afforded by 100-percent redundancy. There is a slight increase in reliability by offering double redundancy rather than single redundancy. The initial weight is doubled by 100-percent single redundancy and tripled by 100-percent double redundancy. Obviously, the number of spares for each item is limited by weight and volume considerations, so the maximum reliability for the minimum weight is desired.




A




B

FIGURE 8-6 RELIABILITY REDUNDANCY

Redundancy can be specified in two ways. The equipment may be designed in such a manner that circuit logic locates and replaces the failed unit with its spare. On the other hand, the failure may be detected by the equipment through self tests and the defective module removed and replaced by man.

It is suggested that approximately 15-percent redundancy be built into the critical circuits themselves and approximately 100-percent redundancy be carried out by modular replacement.

Figure G-7 shows the basic reliability blocks for the telemetry-communications system. The improvement of reliability with man in the loop is based upon 100-percent modular replacement with the defective modules being repaired where possible during noncritical periods.

d. Reliability Program

In order to achieve the required reliability for the telemetry equipment, a program encompassing six basic points is required. These points are delineated below in anticipated chronological order.

(1) A reliability specification for all equipment will be defined.

(2) Reliability units (POMS) will be allocated for the conceptual hardware and for man (see paragraph c.3b.

(3) The reliability unit allocations will be reviewed with considerations of redundancy and preventive maintenance.

Figure G-7 - Communications System

| Phase | Signal Conditioning | Decoder | Receivers | Trans-mitters | Storage | Antenna | Total Reliability | Critical Period |
|---|---|---|---|---|---|---|---|---|
| 2 | 0.9896<br>0.9999 | 0.9886<br>0.9999 | 0.9886<br>0.9999 | 0.9886<br>0.9999 | 0.9896<br>0.9999 | 0.9935<br>0.9999 | 0.9400<br>0.9998 | 12 hours |
| 3 | 0.9913<br>0.9999 | 0.9904<br>0.9999 | 0.9904<br>0.9999 | 0.9904<br>0.9999 | 0.9904<br>0.9999 | 0.9953<br>0.9999 | 0.9500<br>0.9998 | 8 hours |
| 4 | 0.9931<br>0.9999 | 0.9921<br>0.9999 | 0.9921<br>0.9999 | 0.9921<br>0.9999 | 0.9931<br>0.9999 | 0.9970<br>0.9999 | 0.9600<br>0.9999 | 6 hours |
| 5 | 0.9949<br>0.9999 | 0.9939<br>0.9999 | 0.9939<br>0.9999 | 0.9939<br>0.9999 | 0.9949<br>0.9999 | 0.9987<br>0.9999 | 0.9700<br>0.9999 | 2 hours |

Phase 4

X.XX basic equipment reliability

X.XX reliability with 100 percent redundancy

(4) Assistance will be given during the process of equipment design with respect to the points outlined in(2) and (3). In addition, fabrication controls that will guarantee the optimum operational reliability will be established.

(5) The program and design will be reviewed and appropriate changes will be effected in the points outlined in (1), (2),(3) and (4). This effort will be a continuous activity throughout each phase of this program. Considerations will be given to the optimum compromise with respect to equipment redundance, preventive maintenance, and in-flight maintenance and repair.

The detailed program necessary to achieve the desired reliability for the checkout and telemetry equipment is delineated in the succedding paragraphs and exemplified in figure G-8.

e. Step I - Establishment of a Reliability Specification

Initially, the entire system and system requirements will be reviewed for specific reliability requirements. This information, together with the conceptual design and the expected environmental levels, will be utilized to generate a reliability specification. This specification will be different for each phase with revisions reflecting the most recent definition of environmental conditions and system requirements. For Phase I, the environmental conditions will be estimated from existing data on orbital flight environments. In addition to environmental conditions, this specification will also include, but not be limited to, mean-time-between-failure requirements, safety margins, and maintenance philosophies.

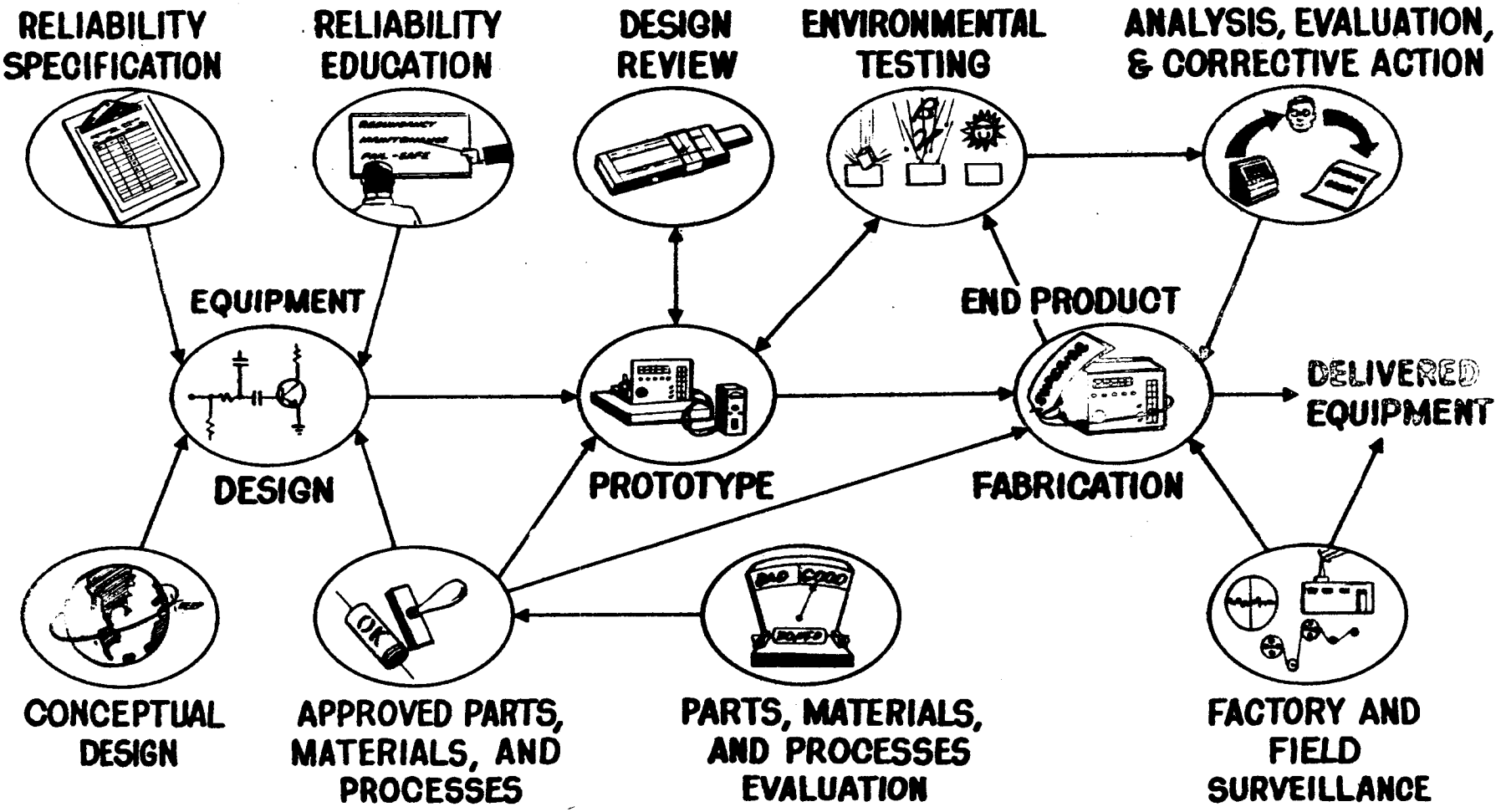



FIGURE G-8 RELIABILITY FOR CHECKOUT TELEMETRY EQUIPMENT

f. Step II - Reliability Education

It is intended to employ reliability education as a means of introducing reliability into this program at the earliest possible date. Initially, the reliability education program will be directed toward the equipment design personnel. It is felt that equipment design personnel versed in reliability techniques and appreciative of the role of reliability techniques in ensuring mission success represent the most desirable foundation for a reliability program. This philosophy coincides with the reasoning that reliability must be built into a product and cannot be tested into it. Practically, this approach will mean a maximum probability of mission success during the early program phases and a minimum amount of design changes throughout the program.

Equipment design personnel will be educated in design techniques stressing the merits of redundancy, self-sensing fail-safe circuitry, self-healing circuitry, and maintenance and considering component application data such as derating, standardization, and failure rates. In addition, reliability prediction techniques will be taught to the equipment design personnel. This phase of the education will allow the designer, to a limited degree, to statistically evaluate problem areas and effect appropriate decisions while compromising equipment redundancies and maintenance without degrading the overall system reliability. While the design personnel have this statistical capability, this does not relieve the reliability personnel from the responsibility of reliability unit allocation.

After equipment configuration has been established, the reliability education of fabrication and test personnel

will be effected. These personnel will indoctrinated into the proper use of a comprehensive failure reporting system and in any special assembly techniques, processes, and tests which are currently nonstandard.

g. Step III - Approved Components, Materials, and Processes

Components, materials, and processes representing the respective state-of-the-art will be assessed relative to applicability, reliability, and availability. To aid in this assessment, information generated from current and future defense and scientific programs will be analyzed; if information is not readily available, special exploratory tests will be performed to obtain preliminary design information. From this industry-wide assessment, a tentative list of reliability approved components, materials, and processes will be generated. Subsequently, equipment design personnel will select their basic design items from this list. Those items selected will then undergo extensive accelerated qualification testing to ascertain mode of failure, mean-time-between-failure, upper and lower bounds of reliable operation, parameter distributions as items are received from the vendor, and parameter distribution shifts with respect to time. The information gathered will be available in sufficient time for the design changes deemed necessary as a result of this program to be incorporated along with the changes deemed necessary as a result of prototype equipment test.

In the event components, materials and processes which are necessary to satisfy the pre-defined reliability requirements are not available, an accelerated developmental program to advance the state-of-the-art will be proposed. If this approach is deemed necessary, results from it will not be available for system integration until Phase II.

Once approved components, materials, and processes have been selected, it will be necessary to ensure that the respective product vendors maintain high quality standards. This will be accomplished by continual requalification testing.

Special treatment will be given to electronic component parts procured for this program. Every electronic component part will be serial numbered so that a detailed history of each part is readily available. The history will be maintained on automatic data handling cards and will include such information as date of manufacture, date of shipment, date placed in stock, date of fabrication into equipment, and date of failure if applicable, and will reflect the parametric values obtained at each point where electrical parameters are measured. These cards will be particularly useful in analysis of data from qualification and re-qualification testing; particularly, a statistical prediction of expected electrical parameter variations with respect to time will be made. Subsequently, the parameter variation versus time studies will be used to establish subassembly failure criteria with respect to operating time and tolerance buildup. In addition, all electrical components will undergo an electrical aging period prior to fabrication into the equipment. The components will be electrically operated for a predetermined period of time to permit detection of all latent manufacturing defects prior to the component's being installed in the equipment. It is necessary that component failures must be held to the absolute minimum in field usage, it is also important that rework (repair) resulting from component failure prior to equipment delivery must be minimized.

This rework, though it makes the equipment operable, increases the probability of a fabrication error being introduced into the equipment and is therefore required to be minimized. Consequently, the electrical aging process is deemed mandatory.

All components, materials, and processes which exhibit a failure at any stage in the program (this includes component manufacture) will be thoroughly analyzed to determine cause of failure. This information will be used to detect potential problem areas and to determine suitable corrective action to prevent the failure recurrence whatever the cause may be

All fabrication processes whether they be standard industrial practices or advancements in the state-of-the-art will be reviewed and evaluated for reliability adequacy with respect to this program. Those fabrication processes found superior will be selected and made mandatory for use on this program.

Finally, by utilizing a selected group of components, materials, and processes and by limiting the equipment design and build to this group, standarization of the program will be effected.

h. Step IV - Design Reviews

A design review program will be utilized to promote design excellence and will be implemented at the initial design phase (where it has the greatest effectivity) at the completion of prototype tests prior to fabrication of deliverable hardware, and to a lesser degree at any point where a design modification is effected. The design

review effort is not designed to optimize design reliability, maintainability, or standardization, for these factors will be an integral part of the equipment design as a result of the reliability education program described previously. Rather, the design review effort will be used to detect problem areas and effect suitable corrective action; such areas of activity might be where the conceptual design was based on an unknown (assumed) parameter, where an initial design requirement is no longer valid, or where the use of a component, material, or process is no longer acceptable. In addition, design review personnel will consider tolerance limits and correlate these limits with the expected component parameter variations with respect to time as established in paragraph c-3g Step III - Approval Components, Materials, and Processes.

The design review activity will be a continuous effort through all phases of this program; however, it will have its greatest contribution for Phases I, II, and III.

i. Step V - Reliability Testing

A reliability test program to verify quantitatively the ability of the system and/or systems to satisfy the number one design requirement, reliability will be conducted. To accomplish this verification, the reliability tests will be designed and conducted at the breadboard and prototype equipment stages. These tests will be effected under multiple environmental levels at the subassembly, assembly, and system level as well as operating life tests under the critical environment. Test plans will encompass Latin Square repeated-test-to-failure techniques to obtain maximum information for minimum cost. This step in the reliability

program will be emphasized during Phases II and III and de-emphasized in subsequent phases as the configuration of the equipment has been established and proven.

Test techniques and philosophies are discussed in detail in paragraph G-4 and G-5.

h. Step VI - Processes and Fabrication Controls

The success of any reliability program is reflected in the operating reliability of the delivered equipment. In order to maximize the ratio of operating reliability to inherent reliability, it is necessary to ensure high quality workmanship, thereby minimizing the equipment degradation introduced during fabrication. This will be accomplished by a detailed program designed to define and effect controls with respect to the fabrication process. Specifically, the following areas will be considered:

(1) Operator Training

As stated previously, high quality workmanship in all stages of the fabrication process is mandatory for achievement of the reliability goals. It cannot be over emphasized that quality must be established and maintained, beginning with the assembly operator or technician who must be thoroughly aware of acceptable quality standards and then be continuously critical of his work. This awareness will be achieved through a training program designed to inform the trainee of necessary fabrication techniques and acceptable fabrication principles. The training program will be culminated when it is certified the trainee is conscious of the quality level required and has developed

proficiency in performing high quality fabrication. Operator or technician self appraisal will be achieved by including in the aforementioned training program a course psychologically designed to instill pride of workmanship and a sense of responsibility through the demonstration of the role played by the system, assembly, sub-assembly, and component part and the effect any failure may have. Subsequent to this initial training and certification program, the control process will consist of operator progress monitoring, period recertification, and supplementary training as processes or fabrication standards are revised. This training activity will be conducted as a phase of Step II, Reliability Education paragraph c-3f.

(2) Operation Instructions and Test Specifications

It is envisioned that some assemblies and subassemblies will require detailed operation instructions for fabrication and test specifications for inspection reliability will review these documents for adequacy with respect to critical processes and/or controls. The "Operation Sheet" procedure as presently employed will provide a means of re-emphasizing special and unique reliability requirements; that is, reliability standards and bulletins for special fabrication processes will be referenced in the Operation Sheets. Test specifications will naturally be reviewed for measurement requirements of the critical parameters. In addition, parameter tolerances will be reviewed for consistency with the results of the serialized component parts control system (Step III) and the design review effort (Step IV).

(3) Special Processes.

In a program where extremely high levels of quality and reliability are established goals, it may be necessary to employ special processes for all or portions of the equipment. An example of such a special process is the aging of all electrical components as outlined in paragraph G-3g Step III, Approved Components, Materials, and Processes. In addition, it is feasible that the equipment design may require that a component, sub-assembly, or assembly be kept absolutely free of microscopic foreign particles; such a requirement would dictate clean-room operations and a special sequence of fabrication.

The equipment will be reviewed during the design and "breadboard" testing phases for areas that require special processes. The results of this review will be incorporated into the prototype fabrication phase. This review will continue to a lesser degree throughout all phases of this program and special processes will be initiated as they are deemed necessary.

(4) Process Controls.

To maintain a high degree of operating reliability, the finished product must conform to its design specifications and standards. This conformation can be accomplished only through rigid process control. The method of control will vary with the process requirements. Some of the process controls which will be utilized are in-process visual inspection, 100 percent critical parameter measurement, continual requalification of component parts, subassembly tests-to-failure on a sampling basis, and redundant inspection. The exact definition of the process controls required is limited at this time by the insufficient information concerning the equipment design. All in-process controls will be defined prior to

the prototype fabrication and will be continually reviewed for adequacy throughout all phases of this program.

(5) Special Handling Requirements.

Component parts, sub-assemblies, assemblies, and complete systems will require special handling to ensure that no degradation in reliability is introduced through antiquated handling techniques. All equipment will be reviewed to determine the optimum handling techniques. Physically delicate component parts and assemblies will be protected from physical damage not only during fabrication but also while they are being shipped, stored, tested, handled, and disassembled for inspection and repair. A complete definition of special handling requirements will not be available until all components have been selected, tested, and evaluated and the equipment design has been defined. Carriers, racks, holders, stands, vehicles, and rooms are some equipment handling aspects that will be considered.

(6) Factory and Field Surveillance.

To ensure that all processes and fabrication controls are rigidly adhered to and to verify that a high level of operating reliability is attained, it is necessary to establish and maintain a factory and field surveillance program. Factory surveillance will be accomplished simply by monitoring the fabrication and test processes to verify that all established reliability principles are discriminately employed. Field surveillance will be employed to verify that the equipment performs its intended function under actual environmental conditions; an observer at all ground based installations where operations are being directed will accomplish this task.

## G-4. EVALUATION CRITERIA

At the component, circuit, and subassembly levels, the reliability will be predicted by determining the failure distributions with resulting confidence bounds placed upon the mean-time-to-failure (MTTF) and mean-time-between-failure (MTBF). From the distributions the required burn-in periods and estimates for wear-out time will be established. It will be necessary also to establish criteria by which tolerance limits may be set. These limits must reflect the degradation expected with time, drift in component values, changes due to severe environment, etc. Components and circuits differ in the effect they have upon the output of an assembly or subassembly. This makes it necessary to establish separate tolerance limits for each item. This requirement is further complicated because of the possible interaction of correlation between contributors.

Suppose the output $X_1$ of a circuit or subassembly is a function of several items $X_2$ ..... $X_n$, where the X's may or may not be correlated. The fundamental task is to obtain the optimum weighting to be assigned to the variables $X_2$ ..... $X_n$ in predicting or explaining variation in the output $X_1$. In the majority of cases, the problem can be linearized. That is, values for $B_1$, $B_2$, $B_3$ ..... $B_n$ can be selected in equation (19) to give the best estimate of $X_1$.

$$X_1 = B_1 + \sum_{i=2}^{n} B_i X_i \qquad (19)$$

where

$\widehat{X_1}$ = best estimate of $X_1$

The coefficients $B_i$ in equation (19) usually involve different units of measurement. To avoid this difficulty the X's are converted to a common standard by the transformation

$$z_j = \frac{X_j - \bar{X}_j}{\sigma_j}$$

where

$\bar{X}_j$ = mean of the distribution of the $X_j$'s

$\sigma_j$ = standard deviation of the $X_j$'s

N = no. of X's for any $X_j$

Equation (19) then becomes

$$z_1 = \sum_{i=z}^{n} \beta_i z_i \qquad (20)$$

where, under this transformation, $B_1 = 0$

The function $\frac{\sum(Z_1 - \hat{Z}_1)^2}{N}$ is to be minimized by the proper choice of the B's. Taking the partial derivative of this junction with respect to the B's and solving the resulting set of equations for the B's give

$$\beta_j = (-1)^j \frac{D_{1j}}{D_{11}} \qquad (21)$$

where

$$D = \begin{vmatrix} 1 & r_{12} & r_{13} & \cdots & r_{1n} \\ r_{12} & 1 & r_{23} & & r_{2n} \\ r_{13} & r_{23} & 1 & \cdots & \\ \cdot & & & & \\ \cdot & & & & \\ \cdot & & & & \\ \cdot & & & & \\ r_{1n} & r_{2n} & & & 1 \end{vmatrix}$$

$D_{11}$ = minor which remains when the first row and first column are deleted.

$D_{1j}$ = minor which remains when the first row and the jth column are deleted.

$r_{ij}$ = correlation coefficient for $z_i$, $z_j$

The B coefficient in equation (19) cannot be used to indicate the relative importance of the variables $X_2 \ldots X_{n1}$ as contributors to the variation of $X_1$. It can be shown however, that the

$$\text{variance of } \hat{z}_1 = \beta_2^2 + \beta_3^2 + \ldots \beta_n^2 + 2 \sum_{\substack{i=2 \\ j=2 \\ i \neq j}}^{n} \beta_i \beta_j r_{ij}$$

The importance of the variable $X_1 \ldots X_n$ can be judged by the magnitude of the squares of the $\beta$ coefficients and the correlation between the variables; hence the tolerance limits can also be established upon the same basis. The correlation coefficients $r_{ij}$ in equation (21) will be established by changing the X's in the breadboard designs by appropriate sampling techniques.

At the assembly level the reliability predictions will be based upon the following Latin Square design. The size of the square will vary with the assembly being tested, ranging from a 3x3 to a 5x5 design. This prediction technique has been successfully demonstrated in present test-to-failure-programs.

Consider a sample of m pieces of equipment (m items) available for testing. Let each item be tested at each of m different stress levels. At each stress level, the item is repaired and retested until r failures have occurred, thus making a total of rm failures for each item. Let j = the number assigned to an item and let $S_L$ be the Lth stress level, where lower subscripts represent less severe levels. Let the order to which the stress levels are applied to the jth item be according to the jth row of an m x m Latin Square. To be specific, let m = 3 and consider the arrangement given in figure G-9.

Figure G-9
Latin Square - Sample of Three Items

| Item No. | Order of Stress Application | | |
|---|---|---|---|
| | 1 | 2 | 3 |
| 1 | $S_1$ | $S_2$ | $S_3$ |
| 2 | $S_3$ | $S_1$ | $S_2$ |
| 3 | $S_2$ | $S_3$ | $S_1$ |

Item No. 3 is subjected to stress level No. 2 ($S_2$) until r failures are observed, then to stress level No. 3 ($S_3$) for r failures, and finally to level No. 1 for r $S_1$ failures, the item being repaired and returned to service after each failure. The number of failures which can be sustained by a single item will depend both on the severity of the environment and the type of equipment being tested.

The equipment has a different mean life, $\theta_L^K$, for each stress level, $S_L$, at which it is tested. Let

$\theta_{jLq}^K$ = the estimate of $\theta_L^K$ for assembly j, where the subscript q indicates the order in which the Lth stress is applied to the item.

An estimate for the mean life can be written

$$\overset{K}{\underset{jLq}{}} = \frac{T_{jq}^K - T_{jq-1}^K}{2} \qquad (22)$$

where $T_{jq}$ = the total time for which the jth has been stressed at the qth stress level applied, and at all previous stress levels.

If the mean life is estimated correctly for each stress level (i.e., if K is chosen properly), then the estimated mean life for a given stress level will be approximately the same whether or not the item had been previously tested. If K is too large, $\theta_L$ will be overestimated; if K is too small, $\theta_L$ will be underestimated. Where age or wear-out

effects have been successfully compensated for, there will be no significant differences between the average values for each of the columns in figure G-10.

The proper value of K can be determined by an analysis of variance, figure G-10, in which the natural logarithm of the mean life is used in the Latin Square.

Figure G-10
Analysis of Variance

| Source of Variation | Degrees of Freedom | Statistical Test |
|---|---|---|
| *S.S. due to age effects = $SS_A$ | 2 | $\frac{SS_A}{SS_E}$ F (2, 2) |
| S.S. due to stress effects = $SS_S$ | 2 | $\frac{SS_A}{SS_E}$ F (2, 2) |
| S.S. due to difference between items = $SS_D$ | 2 | $\frac{SS_D}{SS_E}$ F (2, 2) |
| Error SS = $SS_E$ | 2 | |
| Total SS = $SS_T$ | | |

*S.S. = Sum of squares

The mean life can be approximated by

$$\theta_L^K = (as + b) \qquad (23)$$

where $s$ = stress level

$a, b$ = constants

Independent estimates for a and b can be obtained by standard linear regression theory or by expanding equation (22) in a Taylor's series about estimates for a and b. These estimates are corrected by the resulting values and the expansion repeated. Iteration, continued until the process converges, results in the final estimates.

Once $\hat{a}$ and $\hat{b}$ are determined the reliability is given by

$$\text{reliability} = \epsilon^{-T^{K-(\hat{a}s+\hat{b})}} \quad \text{for } T > 0 \qquad (24)$$

The average strength is defined as that stress level, $s$, for which the reliability in equation (24) is 0.5, and is given implicitly by

$$0.5 = \epsilon^{-T^{K-(\hat{a}s+\hat{b})}} \qquad (25)$$

The reliability boundary, B, is the extreme stress level sustained by the equipment under service conditions. The safety margin, defined as the number of standard deviations which separate the average strength from the reliability boundary, is

$$\text{safety margin} = \frac{K \, \text{LnT} - \text{Ln}\,(-\text{Ln}\ 0.5) - \hat{b}}{\hat{a}} \Big/ \frac{\sqrt{3}\,\sigma}{\hat{a}} \qquad (26)$$

where $\sigma^2$ = residual variance arising from the estimates, $\hat{a}$ and $\hat{b}$.

The maximum safe operating level (MSOL), defined as that stress level for which the reliability of an item tested for a period of time, T, is say 0.95, is given implicitly by

$$0.95 = \epsilon^{-T^K \epsilon^{-(\hat{a}s + \hat{b})}} \qquad (27)$$

The 95 percent lower confidence bound for the MSOL is given by

$$T = \left\{ (-\text{Ln}\ 0.95)\ \epsilon^{\left[\hat{a}\,s + \hat{b} - t_{0.95}\,h\,(s)\right]} \right\}^{1/K} \qquad (28)$$

where

$t_{0.95}$ = value in the "t" tables at the 0.95 level with one degree of freedom

$$h\,(s) = \left[ \sigma^2 \,(C_o + C1\ S_L + C_2\ S_L^2) \right]^{1/2} \qquad (29)$$

where

$$C_o = \frac{\sum_{L=1}^{3} S_L^2}{3 \sum_{L=1}^{3} (S_L - \bar{S})^2}$$

$$C_1 = \frac{-2\bar{S}}{\sum_{L=1}^{3} (S_L - \bar{S})^2}$$

$$C_2 = \sum_{L=1}^{3} (S_L - \bar{S})^2$$

If the stresses during flight, $\psi$, are expressed as a function of time, then reliability (flight) =

$$\int_{t=0}^{T} e^{-(\hat{a}\psi + \hat{b})} kt^{k-1} dt \qquad (30)$$

where T = time of flight.

Estimates for the stresses, $\psi$, will be determined by preliminary studies, tests and telemetry data. With present information it is felt that the temperature, vibration, and radiation environments can be treated independently in the Latin Square analysis. The combination of the results of all these environments will then constitute a basis for the flight reliability of the checkout and telemetry systems.

## G-5. PROBLEM AREAS

### a. Man's Maintenance Capability

One problem area, the extent of which is still undefined, is man's effectivity in performing maintenance activity in a space environment. Man's capabilities and limitations in a space environment will only be defined after the completion of a space mission. There are, however, certain basic assumptions that can be made at the present time. The assumptions are delineated below:

(1) Man will be able to perform certain maintenance and operating functions on the OLF and S/C.

(2) In the absence of a gravitational environment and mechanical restraints, the average maximum continuous hand force which a man can exert is equivalent to a torque of 2.5 foot-pounds. With mechanical restraints this capability is considerably improved.

(3) Man's (unit)proficiency index will improve with time and experience in a fashion similar to that depicted graphically in figure G-11.

(4) Man's ability to perform maintenance while being a member of a three man crew will not be decreased by the loss of one crew member.

FIGURE G-11 UNIT PROFICIENCY GROWTH

The proposed approach to optimizing reliability is to have a built-in circuit redundancy of approximately 15 percent (see figure G-3) and 100 percent spare modules. The proposed approach to compromise for a loss in man's effective maintenance is to have small, light-weight, modular replacement units, and the capability of a shift upwards in the built-in circuit redundancy requirements. Thus, the unanswered question of the degree of man's effectivity in space imposes an additional design requirement: namely, the basic design of the telemetry and communications equipment must be flexible to the degree that built-in redundancy can be increased if required. Consequently, it is proposed that a continuous study program for evaluating and optimizing man's effectivity in space be included as a portion of the reliability effort.

b. Long-Term Component Stability. For many of the components which are to be used in this operation, long-term stability characteristics as functions of environment are undefined. In cases where long-term component stability information is presently available, it is felt this information is not compatible with the anticipated environments for the OLF and S/C. Since many of the environmental factors contributing to long-term component stability requirements are undefined for the OLO mission, it is proposed that a study program on this problem be effected. This study program would define critical component stability, and in conjunction with Step III, Approved Components, Materials, and Processes, paragraph C-3g, approved components will be evaluated for stability characteristics. In addition, this study program will investigate accelerated methods of duplicating long-term environmental

exposure. The end result of this program will be revised maintenance techniques designed to replace equipment before degradation failures are experienced.

c. Optimizing Reliability Techniques

A study program to develop and optimize reliability test techniques must be initiated at the inception and continued throughout all phases of this program. Some predictable factors necessitating this study program are design changes in the equipment, improvements in the state-of-the-art in environmental testing, and continual experience gained in testing. In particular, some of the study areas to be considered are

(1) Formulation of statistical models for all equipment operating modes with emphasis on equipment degradation. These models will consider man as an integral part of the equipment.

(2) Establishment of a comprehensive equipment analysis and tolerance study (Step IV, Design Reviews, paragraph C-3h) to optimize reliability unit allocation as the design activity progresses.

(3) Revisions of the data acquisition system to permit effective use of computer-program failure and test information on such a time scale to allow resultant corrective action to be integrated into the equipment within the over-all program schedule.

(4) Modifications to existing reliability test techniques to employ improvements in the state-of-the-art of environmental testing and to utilize cumulative test experience gained during this program.

## G-6. CONCLUSIONS

It has been concluded that the reliability requirements for telemetry and communications can be met consistent with the program schedule. The reliability goals can be achieved without a significant state-of-the-art improvement by the use of critical circuit redundancy, one complete set of spares for maintenance, and the use of man to perform maintenance.

a. The use of man as a maintenance agent is the optimum approach for attaining high reliability. Complete redundancy (100 percent spares) is proposed because of the low weight of the conceptual design and a balance of yield with cost-weight factors.

b. Any improvement in components and circuitry techniques enhancing reliability should be utilized to effect a longer mean-time-between-failure rather than to reduce the number of spare modules.

c. The anticipated vibration and temperature environment does not present any special problems with present design techniques and components. The radiation environment and its effect on long-term component stability cannot be defined; however, it is assumed that if man can survive, the equipment will survive though its operating capability may be degraded.

d. Component development does not appear to be an area for concentrated reliability improvement; the normal component state-of-the-art improvements are commensurate with the over-all program goals.

## G-7. ASSUMPTIONS

The assumptions used in the definition of the OLO reliability program are as follows:

a. The basic design of the ground, OLF, and S/C telemetry and communication equipment is compatible. Advantages resulting from this compatibility are standardization and increased confidence from a larger sample size.

b. Man will be capable of performing certain maintenance and operating functions on the S/C and OLF.

c. Man's ability to perform maintenance while being a member of a three man crew will not be decreased by the loss of one crew member.

d. For reliability test and analysis purposes, the environments are considered to be independent.

e. As a compromise between reliability and maintenance requirements, "Meca" packaging concepts and welded circuits will be used.

f. The anticipated vibration and temperature environments do not present any special problems with present design techniques and components.

g. Prototype equipment will be available for reliability testing.

## G-8. RECOMMENDATIONS

It is recommended that additional study be conducted in the areas delineated and that the reliability program as outlined in paragraph G-3, Steps I through VI, be utilized to meet the reliability requirements.

# APPENDIX H

# CONCEPT OF MAINTENANCE AND LOGISTICS

## H-1. INTRODUCTION

The present state-of-the-art of component technology and system design for reliable operation is not sufficiently advanced to field a system capable of executing the OLO mission without a system failure. Therefore, in order to achieve the desired probability of mission success, methods of supplementing the reliability effort must be examined. One logical method is provided by appropriate maintenance or repair of the equipment used in the operation. Maintenance will result in an optimum system when a trade-off between reliability of mission success and costs is considered.

Maintenance, however, implies replacement of malfunctioning items and creates a logistics or re-supply problem. Both maintenance and logistics as applied to OLO will be discussed in this Appendix.

## H-2. OBJECTIVES

The objectives of this Appendix are to examine the maintenance and logistics problems associated with the OLO program. Included in the discussions will be:

a. Definitions of maintenance and logistics

b. A discussion of maintenance requirements

c. A discussion of maintenance methods

d. A discussion of design concepts for maintenance

e. A discussion of man's role in maintenance

f. A discussion of stockage and re-supply of spare parts.

## H-3. APPROACH

In essence, "maintainability" denotes the amenability of a system to repair or maintenance which will ensure that a system has a high probability of successfully completing its mission. In-flight or orbital maintenance may be divided into two areas:

a. Testing and failure diagnosis

b. Corrective or other actions (includes alignment, calibration, repair, or interchange of equipment).

The denoted functions can be performed manually or with automatic test equipment, but in either case equipment, personnel, facilities, procedures, and time to accomplish the desired result are required.

Maintainability remains a basic necessity because equipment used in space missions must operate reliably with exacting precision during long-term mission under severe environmental conditions. These requirements suggest a highly integrated, complex system that must have a mean time to failure (MTF) much greater than the time required

to complete the mission. It is significant to note that maintenance must be considered as a basic design criterion from the beginning of the program. If it is considered early it will create a system which may be maintained easily and reliably. If maintenance requirements are ignored until late in the program, the penalties of cost and schedule delays may prohibit the inclusion of an optimum maintenance capability and result in a decrease in probability of mission success.

a. Maintenance Methods

(1) Maintenance Model. A model of the steps required for the performance of maintenance is illustrated in the figure H-1:

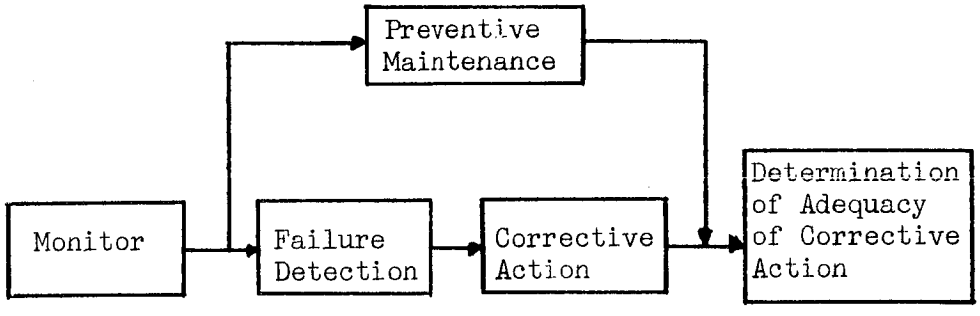



Figure H-1. Maintenance Model

The model holds true for automatic, semiautomatic, or manual maintenance. The fully automatic system would automatically test itself and then monitor the functions in the rest of the system. An alternate method would bypass the self-test feature and use a majority-of-dissenting-vote technique for failure detection. After a failure is detected, an automatic repair or self-healing technique would be used. This

will consist of switching to an alternate signal flow path or some equivalent operation. After the new flow path has been established, a verification test will be made to determine if the equipment has been restored to a normal mode of operation. Preventive maintenance may be performed by apparatus sensing the operating hours or rate of degradation and switching to an alternate path.

Manual maintenance is accomplished by reading of meters, scopes, and other test equipment by a human operator. Corrective action is taken by the operator's manually switching to alternate equipment or replacing of assemblies. The operator may also perform preventive maintenance by replacing equipment where degradation is indicated or when the MTTF of the equipment has been reached.

The semiautomatic maintenance scheme would combine elements of the automatic and manual maintenance schemes in an optimum manner: full automation is required in areas where repair time must be held to a minimum, but manual operation is used where flexibility is required and time permits. This system is the most practical of those presented and will be adopted if further study substantiates the advantages given above.

(2) Repair Techniques. Once a failure is detected, either an adjustment or a repair by a replacement module is required. The use of digital equipment will do much to decrease the number of adjustments, but some of the analog equipment may require adjustment. Due to a logistics consideration, an adjustment is preferable to a replacement and scrapping of the item and will therefore be used as a repair technique.

Replacement of a malfunctioning part may be accomplished by modular or piece part replacement. Piece part replacement is preferable from the standpoint of minimum weight and cost. However, the complexity of test equipment for isolation to piece part level will impose severe weight problems. The time required for piece part replacement and the reduced reliability as a result of semi-skilled operators in an alien environment are factors which prohibit piece part replacement. Dense packaging of electronics may also prevent reliable replacement of piece parts.

Repair will therefore be accomplished by modular replacement. However, the basic tools for piece part replacement will be furnished for use in case of extreme emergencies only. For example, if all operating equipment and spares of one type are inoperative, man may attempt to cannibalize piece parts from one unit to repair another.

Any replacement of piece parts will be done by welding techniques, and attempts will not be made to rebond components to their boards or to replace potting compounds because of problems associated with such operations in a space environment.

b. Design Concepts for Maintenance

(1) Optimum Modular Size. The reliability studies in Appendix G indicate that at least 8 to 16 replaceable assemblies are required for each subsystem to achieve the degree of reliability required for mission success. Maintenance and logistics requirements, however, dictate that a larger number of replaceable modules be used. Additional study will be required to determine the optimum number of

modules and hence the optimum size. Trade-offs in this area will be maintenance capability, logistics requirements, reliability, weight, volume, and cost.

(2) Accessibility. In order to permit repair by replacement in a minimal time, a convenient design must be established to give the operator maximum access to the equipment. This dictates that the equipment must be housed in a "shirt sleeve" environment. Figure H-2 illustrates a method of packaging for optimum modular replacement. This scheme has merit from the standpoint of packaging density, maximum accessibility, and maximum troubleshooting capability. Many other schemes are possible and must be evaluated, however.

(3) Updating of Equipment. During the evolutionary OLO program many changes will be required to portions of the equipment which will have an influence on system communications. To avoid scrapping of equipment already in orbit, a maximum flexibility is required of the equipment. A packaging technique such as shown in figure H-2 will give this flexibility. Modules and subassemblies may be replaced or interchanged to effect a new configuration. It should be noted that actual packaging densities will be greater than those illustrated in figure H-2.

(4) Levels of Maintenance. Two levels of maintenance are possible with the figure H-2 packaging. Subassembly replacement will be accomplished should a failure occur during a mission. Modular replacement will be performed during down time between missions.

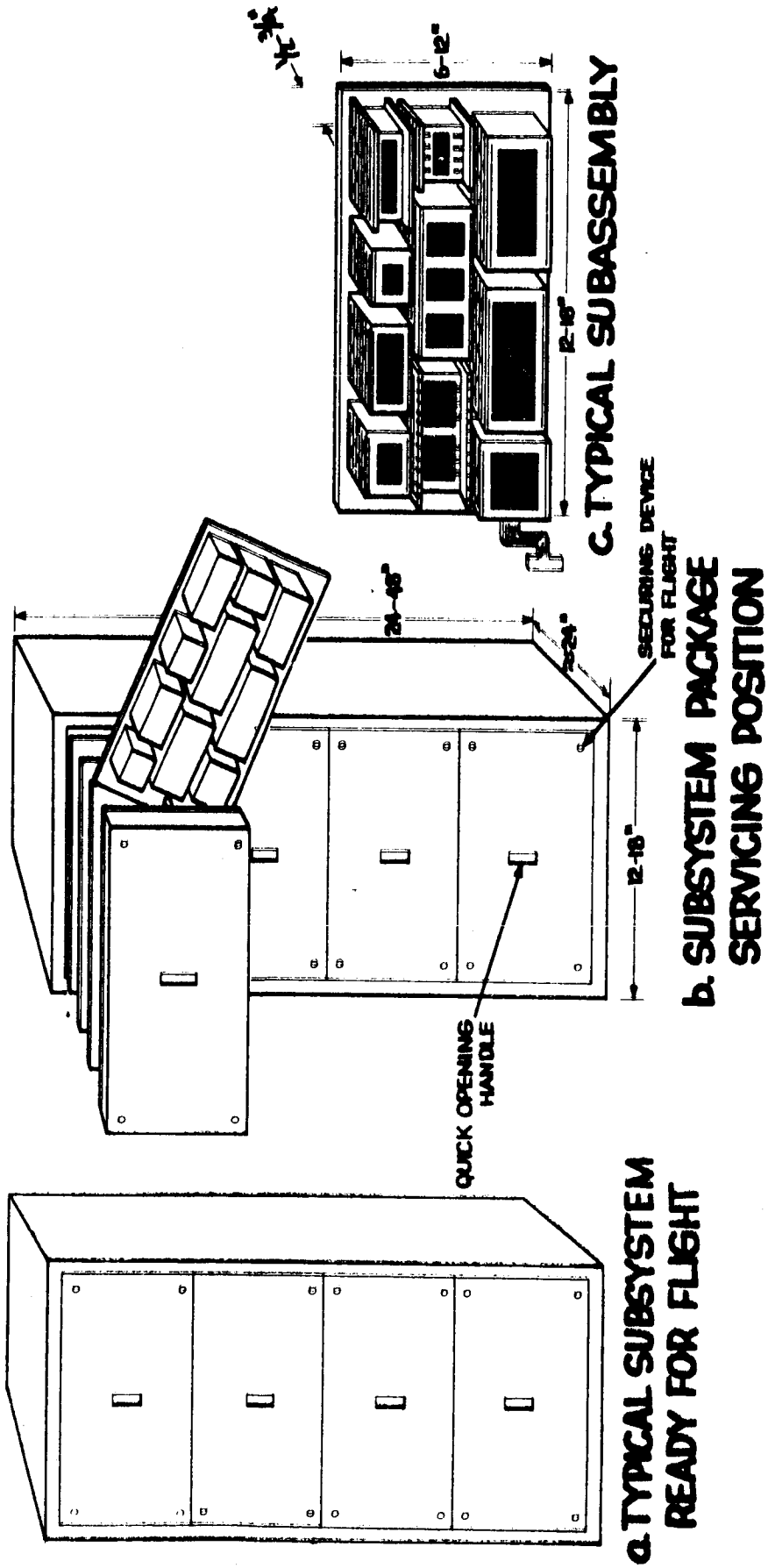



FIGURE H-2 PRELIMINARY PACKAGING CONCEPT

(5) Standardization. Due to weight limitations a concerted effort must be made to standardize in the areas of tools and test equipment. As an example, the same wrench should be used for as many applications as possible, a multimeter may serve many functions, and so on. Close coordination between all contributors to the OLO program must be maintained to effect this discipline.

(6) Maintenance Aids. Since maintenance will be required of the crew, aids must be furnished such as tee holds, handles on equipment, or a strap down seat to enable the operator to exert a sufficient torque or push to complete the maintenance job. If these are not furnished, countermotion of the man's body may prevent completion of the job.

c. Man's Role In Maintenance.

(1) General. It has been pointed out that the more sophisticated maintenance scheme that excludes man will become extremely complex and that the equipment will be prohibitively heavy. The use of man in the maintenance loop is essential, since his capabilities can be substituted for mechanical sophistication.

(2) Problem Areas. The most significant problem concerning the use of man in the maintenance loop is how efficient and stable he will remain in the space environment. Studies have been carried out to determine effects of weightlessness, absence of friction, and spatial disorientation on sensory processes and basic psycho-motor performance, manual application of forces, and handling of basic tools in a space environment. However, additional research must be completed

before man may be used to his best advantage in the OLO program. These studies indicate that man is capable of completing all basic maintenance functions.

Other studies have indicated that the most significant function of man is not necessarily manipulation of hardware, but his intellectual performance as a decision-maker. These studies also point out that human operators, if supplied with a priori reliability information, do "converge upon an optimum or ideal mathematical solution. This would indicate that the human operator trained in decision making could perform adequate maintenance." The problem then is how shall the operator be furnished with the requisite a priori information. This may be accomplished with maintenance aids and proper training.

(3) Operator Maintenance Aids. Two methods of providing the operator with the requisite information are through a microfilm library and through contact with a technical group on the earth. The microfilm projector may be used for recreational reading purposes. The microfilm library should include such items as procedures, functional block diagrams, schematics, spares, lists, and decision tables. This information will be used for unusual problems or in cases where a failure area is not isolated to a specific subassembly or module.

The ground support complex may also aid the operator by serving as a braintrust and directing the trouble-shooting activity when required.

(4) Training of Operators. To be effective the operator must know his equipment well enough to "field strip it while blind folded". To accomplish the required training an operator

must enter the project during the initial design phases and work with design engineers from the program's inception. Provisions will be made to train the operators at the factory and in the field during the development phase.

## H-4. LOGISTICS

a. Definition. Logistics may be defined as the methods employed in the details of transport, storage, and supply of maintenance equipment. Logistics requirements are a direct function of maintenance requirements which in turn are dependent on failure rates.

b. Stocking of Repair Parts. A study is required to determine optimum types and levels of spares to be stocked at various points such as the OLF, launch pad, factory, and lunar base. This basic equipment reliability and repair philosophy must be considered in this study. The best present estimate of minimum spare requirements are 15 to 30 percent for the OLF, 100 percent for both the launch pad and the factory. This study must also include an investigation of storage, indexing, and inventory techniques, access requirements, etc. Storage in racks similar to the one in figure H-2 will satisfy most of the storage access problems.

c. Re-Supply of Spare Parts. A basic assumption used in the reliability studies is that an adequate supply of spares will be available when required, which in turn presents a re-supply problem. The method of replacing expended spares presents a severe economic problem and is an area which

deserves further study. One possible method of replacing critical parts on the OLF at a somewhat reduced cost is to use a smaller, unmanned missile system such as the Blue Scout for small shipments (100 to 200 pounds).

d. Disposal. Disposal of spent parts represents a problem in space. To eject them into orbit would jeopardize future space operations. To send them back to earth requires an unwarranted consumption of fuel. The most feasible solution is space garbage bags suspended outside the OLF for the period of its useful life.

## H-5. EVALUATION CRITERIA

At this point consider some of the factors which have an effect on the maintainability of a system for the purpose of defining a figure of merit or predictor of maintainability. This predictor may be useful in evaluating proposed maintenance systems.

Factors which influence maintainability are as follows:

a. MTTF - Mean Time To Failure - A time dependent function of the reliability of the systems normalized with respect to the time required to complete mission.

b. MTTR - Mean Time to Repair - Normalized with respect to mission time allocated for repair.

c. CTR - Cost to Repair - Normalized with respect to mission cost. Cost of repair includes part and transportation costs.

d. POMS - Probability of Mission Success - The required probability for completing the mission successfully.

e. CORF - Cost of Repair Facility - Includes tools, test equipment, etc. Normalized with respect to mission cost.

f. RATRS - Ratio of Available Spares to Required Spares - One method of combining these factors to yield a figure of merit for maintainability (M) is

$$M = \frac{(MTTF)\ \ (POMS)\ \ (RATRS)}{(MTTR)\ \ (CTR)\ \ (CORF)}$$

The multiplicative mode was chosen on rational grounds because it appeared that a simple additive relationship would not give adequate weighting to any one factor when that factor approached an extreme. It was also assumed that a more complex or second order relationship was not justified.

Perhaps the two most important factors in a space mission are probability of success and cost. Figure H3a illustrates POMS versus mission cost for various values of M. These curves represent trends and, as such, were not generated with actual data.

Figure H3b illustrates the degree to which maintenance can increase the POMS with a resultant program cost savings. This savings is made possible by an initiation of maintenance when the reliability program has entered the asymptotic region. A study will be required to determine where the optimum point (1) on figure H-3b is located. This study must include results of initial reliability and cost studies.

## H-6. PROBLEM AREAS

A number of problem areas requiring resolution have been discussed in previous sections. An outline of maintenance and logistics problems is given in figure H-4.

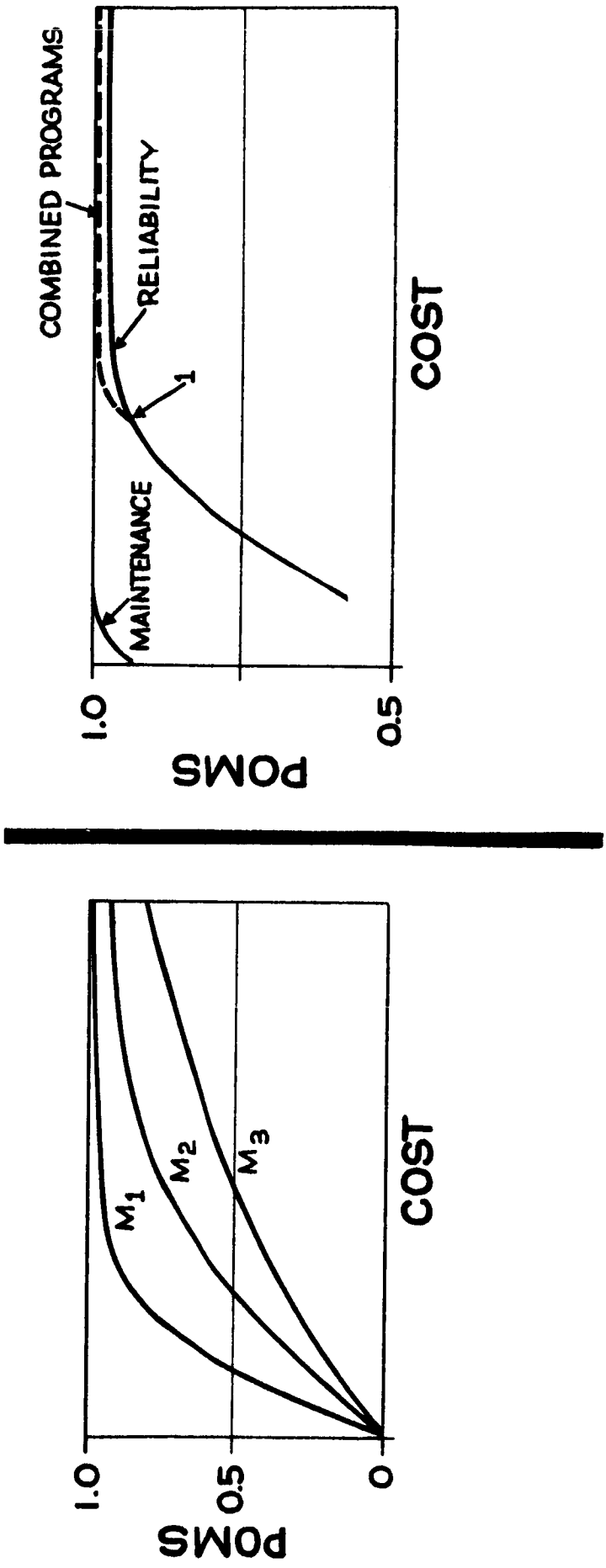



FIGURE H-3 EFFECT OF M& MAINTENANCE ON POMS & COST

FIGURE H-4

Maintenance and Logistics Problems

| Problem | Phase | Statement of Problem | Proposed Solution(s) | Proposed Studies |
|---|---|---|---|---|
| Repair Techniques | All | Methods are required to replace malfunctioning modules, cables, etc., in a space environment. | Welding and Modular replacement. | Study is required to determine optimum techniques. |
| Accessibility | All | Packaging and layout of equipment must be such that access for maintenance is provided. | Include maintenance as a primary design criteria. | Determine optimum packaging and layout criteria. |
| Updating of Equipment | All | Provisions for updating techniques must be devised to prevent obsolescence of equipment. | Include updating provisions in original design. | Study of updating requirements and techniques. |
| Levels of Maintenance | All | The extent of maintenance required under varying conditions must be established. This in turn will influence test equipment requirements, tools, and workload, on the human operator. | Establish trade-offs as required. | Define required levels of maintenance. |

Figure H-4 (Cont)

| Problem | Phase | Statement of Problem | Proposed Solution(s) | Proposed Studies |
|---|---|---|---|---|
| Standardization | All | A major effort is required to standardize test equipment, tools, electronic modules, etc., to create flexibility and reliability and to reduce weight. | Set up program to define requirements and direct action in this area. | Study to determine optimum standardization program. |
| Maintenance Aids | All | The operator must be furnished with aids to adapt him to a space environment and enable him to perform normally. He must also be furnished with data concerning troubleshooting and maintenance. | Define and design aids which will assist the operator in maintenance. Examples are toe and hand holds, microfilm reader with microfilmed procedures, etc. | Study to determine requirements and design requisite equipment. |
| Operator Training | All | Extensive training of operators in the maintenance area must be accomplished. | Set up a program to train the operators. | Study and outline a training program. |
| Re-Supply | All | Maintenance and repair operations require the use of spares. These spares must be replenished occasionally. | Define primary and secondary modes of re-supply. | Investigate methods and cases of re-supply. |

Figure H-4 (Cont)

| Problem | Phase | Statement of Problem | Proposed Solution(s) | Proposed Studies |
|---|---|---|---|---|
| Parts Storage | All | An optimum system must be set up for storage, inventory, and indexing of repair parts. | Many varied systems may be used. A study is required to determine which is optimum. | Determine optimum system. |

## H-7. CONCLUSIONS

Maintenance in space will remain a necessity until the reliability of present equipment is increased by several orders of magnitude. It now appears that man will be capable of performing the required maintenance provided he is furnished with proper maintenance aids. Additional studies are required to determine man's limitations and the maintenance aids he will require. Studies are also required to determine techniques of maintenance in space.

## H-8. RECOMMENDATIONS

It is recommended that man be used in the maintenance function through all phases of the OLO program. It is further recommended that a series of studies be conducted to determine man's capabilities for maintenance and which aids are required for his maintenance mission.

# VOLUME V

## ORBITAL LAUNCH OPERATIONS

## PART II

## ATTITUDE CONTROL

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# SECTION I

## SUMMARY

The Sperry Rand Corporation is supporting the Vought Astronautics study of Orbital Launch Operations (Contract NAS 8-853) by investigating three Astrionics areas. The subject of this volume is Attitude Control of OLO vehicles, which covers the work accomplished by the Air Armament Division of Sperry Gyroscope Company and by Vickers, Inc., both Divisions of Sperry Rand Corporation.

The principal conclusions of this work are:

a. The attitude control aspects of the OLO program are within the time scale for state-of-the-art development as conceived by Sperry Rand. No major technological breakthroughs are necessary to successfully establish orbital launch operations.

b. Certain follow-on study work is absolutely necessary to firmly establish plans for successful orbital launch operations. In the area of attitude control these are:

(1) To establish the relative-attitude sensors and method of relative-attitude control required to operate two OLO vehicles in close proximity to one another.

(2) To prepare specifications for attitude control equipment to be used in the initial orbital-launch facility, the orbital-launch vehicle, and unmanned payloads.

Since the above work is well within the capabilities of the Air Armament Division of Sperry Gyroscope Company and

Vickers, Inc., it is requested that assignment of this work to Sperry Rand for the next funding period be considered.

c. Other recommended follow-on work is as follows. These items, while not placed in the absolutely necessary category, are still considered necessary for well-balanced planning and the proper initiation of work programs necessary to meet the objectives of the national OLO program on schedule.

(1) To proceed with further development of storable bi-propellant attitude controls.

(2) To establish the optimum attitude-control technique for use with a coupled tug/payload combination.

(3) To prepare specifications for the attitude control equipment required in the intermediate orbital-launch facility, the intermediate unmanned payload, and the astro-tug.

These programs are also recommended for Sperry Rand support of the Vought Astronautics OLO Development Program.

d. Some of the major technical conclusions, of interest to Vought Astronautics as an aid in planning the above work, are as follows:

(1) Development of special relative-attitude sensors will be required for use during docking or joining of two vehicles. Otherwise, component requirements are well within the state-of-the-art.

(2) It appears that electrical as well as mechanical interconnection of the astro-tug and the incoming payload will be advantageous from the standpoint of fuel economy and equipment weight savings.

(3) When operating unmanned payloads, it appears that the attitude command function can be performed remotely from the orbiting facility. This should allow a significant reduction in the weight of payload attitude-control equipment.

# SECTION II

# INTRODUCTION

This report presents the results of a preliminary study of attitude-control requirements for the vehicles to be used in Orbital Launch Operations (OLO). The study was conducted for Vought Astronautics by the Research and Advanced Development Directorate of the Air Armament Division of Sperry Gyroscope Company with the assistance of Vickers, Inc. The objectives of the study were to synthesize candidate attitude-control systems for each OLO vehicle, to isolate the basic attitude-control problems peculiar to Orbital Launch Operations, and to recommend programs that will resolve the problems and produce equipment designs suitable for the applications.

The Orbital Launch Operation will consist of placing two or more payloads in an earth orbit, combining all or part of the individual payloads into a space vehicle, and launching this vehicle from orbit. This operation can be performed in several ways. The simplest type of Orbital Launch Operation would proceed in the following manner: unmanned payloads would be boosted into orbit by large, non-manrated boosters and the crew would follow in a smaller, more reliable vehicle. A launch vehicle would be assembled from the several vehicles and injected into a lunar or deep-space trajectory. Nothing of importance would remain in orbit after the launch vehicle departed.

A more sophisticated procedure would be to orbit a large permanent facility, sending the facility crew up after

a satisfactory orbit is assured. Several payloads, both manned and unmanned, would follow and the facility crew would then assemble and/or fuel a launch vehicle. The permanent facility and its crew would remain on station for later operations. An astro-tug might be required for payload retrieval and launch vehicle orientation in the course of this orbital launch operation.

Since the type of Orbital Launch Operation to be employed has not yet been finally established, three basic OLO complexes must be considered: systems without a permanent orbiting facility, systems with a permanent facility and no astro-tug, and systems with both a permanent facility and an astro-tug. An effort was made in this study to compare the overall attitude-control requirements for these three alternate configurations, as well as to synthesize tentative attitude-control systems for the individual vehicles in each complex.

Figure 2-1 shows a matrix of the vehicles required in each of the alternate configurations. The first column of figure 2-1 summarizes the several attitude-control schemes considered for each vehicle. The control methods are manual control of remote vehicles from the permanent facility, automatic control of remote vehicles from the permanent facility, self-contained manual control of manned vehicles, and self-contained automatic control of unmanned vehicles. The applicability of each method varies, of course, with the vehicle and with the phase of operations in which the vehicle is employed.

The remaining columns of figure 2-1 list phases of the Orbital Launch Operation in which attitude control of one or more vehicles is required. These include boost to earth

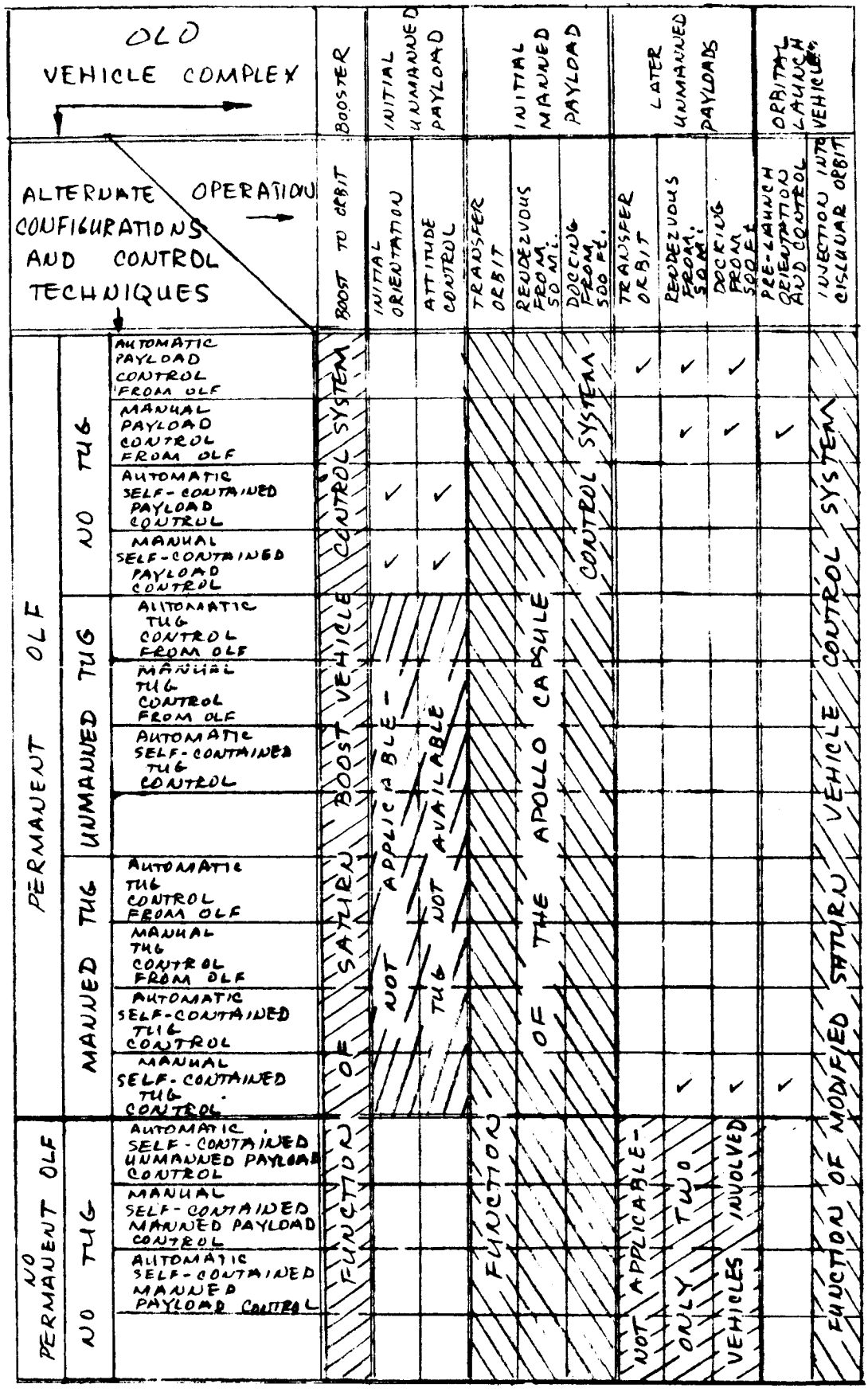



FIGURE 2-1 ALTERNATE CONFIGURATION MATRIX

orbit, rendezvous of various vehicles, and cislunar injection of a launch vehicle. As noted in the figure, boost to orbit, rendezvous by manned payloads, and cislunar injection will be performed by modified Saturn or Apollo vehicles. Attitude control systems are already available for these vehicles, and were therefore not considered in this study. For the purposes of this discussion, Orbital Launch Operations begin when the last booster stage separates from an unmanned payload and end when the main propulsion rocket on the orbital launch vehicle is ignited.

The program time schedule made investigation of all control schemes in every vehicle impractical. Accordingly, the checked combinations of Figure 2-1 were selected for detailed study. At least one example of every control scheme under consideration is included.

The areas investigated in synthesizing attitude control systems for these vehicles were:

- Manual and/or automatic attitude and vernier-velocity controls
- Stability augmentation
- Reaction-control torquers and vernier-velocity thrusters.

Guidance and navigation equipments were considered only to the extent necessary to define guidance/control-system interface requirements. Particular attention was given to the role of the human in performing attitude control, and an effort was made to establish which functions are best suited to manual control. Detailed study of such items as manual override and authority limits for automatic systems, automatic trimming of manual systems, and synchronization of

the automatic and manual systems were reserved for follow-on programs.

In performing trade-offs, reliability was considered of primary importance and, subject to performance limitations, was given highest priority. Weight was also given considerable attention, particularly when equipment was to be duplicated in every payload boosted up from earth. An attempt was made to centralize common equipments in the permanent facility whenever possible.

The following sections of this report summarize the study and its findings. Section III describes the vehicles, their missions, and their attitude control requirements. Comparisons of the permanent facility/no permanent facility and tug/no tug configurations are drawn directly from this basic information. Section IV discusses the control systems recommended for each vehicle, together with alternate systems, and trade-off considerations. Section V summarizes the program cast and schedule through 1966, while Section VI is devoted to detailed discussion of the program proposed for calendar year 1962. Finally, Section VII summarizes the recommended action to be taken in developing each of the vehicle attitude-control systems.

# SECTION III

# INITIAL CONSIDERATIONS

## 3-1. VEHICLE CHARACTERISTICS, MISSIONS, AND ATTITUDE CONTROL REQUIREMENTS

The vehicles described below are a composite of those being considered by Vought Astronautics at the time this study was initiated. The quoted performance specifications are based on documents supplied by Vought Astronautics, extractions from the literature, and data obtained from other space programs underway at Sperry. Accordingly, the vehicle and mission characteristics, and attitude control requirements, are of a preliminary nature and will be updated as more definitive information becomes available in follow-on programs.

### a. Orbital Launch Facility

A sketch of the Orbital Launch Facility (OLF) and a tabulation of its attitude control requirements are shown in figure 3-1. The vehicle consists of a hangar, living quarters, airlock with attached escape vehicle and Astrotug, and a nuclear reactor. The attitude control functions include initial alignment, coarse attitude hold, orientation for docking, and fine attitude hold.

The OLF is considered to be boosted into orbit with its longitudinal axis approximately horizontal. The first function of the attitude control system is to stop

any initial tumbling and rotate the vehicle until its longitudinal axis is vertical. Vertical orientation was selected to take advantage of the restoring torque due to the gravity gradient. The time allotted to damp a 1-rpm tumbling rate is 9 minutes if the tumbling is around the yaw axis, and 90 minutes if the tumbling appears in pitch or roll. In reality, pitch tumbling rates will be much lower than yaw tumbling rates because of the different moments of inertia.

The attitude control system must maintain the longitudinal axis of the OLF within 5° of the local vertical when the coarse-attitude-hold mode is engaged. In addition, the OLF may have to be reoriented for docking of an incoming payload, and will be required to maintain the selected docking orientation very accurately. A docking reorientation maneuver, consisting of ±180° rotation about the local vertical and up to 30° departure from the local vertical, has been specified. The 30° limit allows some simplifications in the attitude control system. A fine-attitude-hold capability of ±0.5° has been specified for use during docking operations.

The positioning-accuracy and threshold-rate specifications for this vehicle were obtained from documents supplied by Vought Astronautics. The remaining quantities could then be computed. As an aside, some adjustments are acceptable in the quoted torque specifications. Generally speaking, the maximum torque level can be varied if the torque-time product remains constant. However, the control system must always meet the threshold-impulse and minimum-pulse-width specifications.

| REQUIREMENT \ FUNCTION | INITIAL ALIGNMENT | | ATTITUDE HOLD | | ORIENTATION FOR DOCKING | | DOCKING ATTITUDE HOLD | | |
|---|---|---|---|---|---|---|---|---|---|
| ATTITUDE CONTROL | PITCH YAW | ROLL | PITCH YAW | ROLL | PITCH YAW | ROLL | PITCH YAW | ROLL | |
| 1. POSITIONING ACCURACY | ±5° | ±5° | ±5° | ±5° | - | - | ±.5 | ±.5 | DEG |
| 2. MAX. ACCELERATION | $10^{-3}$ | $10^{-2}$ | - | - | $10^{-3}$ | $10^{-2}$ | - | - | DEG./SEC.$^2$ |
| 3. THRESHOLD RATE | .05 | .05 | .05 | .05 | - | - | .05 | .05 | DEG./SEC. |
| 4. STEADY DISTURBANCES | - | - | .045 | - | - | - | 5 | - | FT.-LBS. |
| 5. VARIABLE DISTURBANCES | - | - | .045 | - | - | - | .045 | - | FT.-LBS. |
| 6. MAX. DEMAND, STEADY | 58 | 52 | - | - | 52 | 52 | - | - | FT.-LBS. |
| 7. DURATION MAX. DEMAND | 270 | 540 | - | - | 270 | 540 | - | - | SEC. (REPEATED) |
| 8. THRESHOLD IMPULSE | $3x10^3$ | $3x10^2$ | $3x10^3$ | $3x10^2$ | - | - | $3x10^3$ | $3x10^2$ | FT.-LB-SEC. |
| 9. MINIMUM PULSE-WIDTH | 0.5 | 0.5 | 0.5 | 0.5 | 0.5 | 0.5 | 0.5 | 0.5 | SEC. |
| THRUST CONTROL | | | | | | | | | |
| 10. MAX. FORE/AFT. ΔV | — | | — | | — | | — | | |
| 11. MAX. LATERAL ΔV | — | | — | | — | | — | | |
| 12. THRESHOLD ΔV | — | | — | | — | | — | | |
| 13. MAX. DEMAND, STEADY | — | | — | | — | | — | | |
| 14. DURATION MAX. DEMAND | — | | — | | — | | — | | |
| 15. THRESHOLD IMPULSE | — | | — | | — | | — | | |

INITIAL ALIGNMENT

DAMPING IRPM INITIAL TUMBLING RATE: 90 MINUTES
ROTATE 90° IN PITCH: 9 MINUTES IF NO TUMBLING

ATTITUDE HOLD

SALVE TO LOCAL VERTICAL

ORIENTATION FOR DOCKING

ROTATION OF 360° MAX AROUND LOCAL VERTICAL
±30° DISPLACEMENT FROM LOCAL VERTICAL

DOCKING ATTITUDE HOLD

MAINTAIN SELECTED ATTITUDE FOR 20 MIN. DOCKING MANEUVER

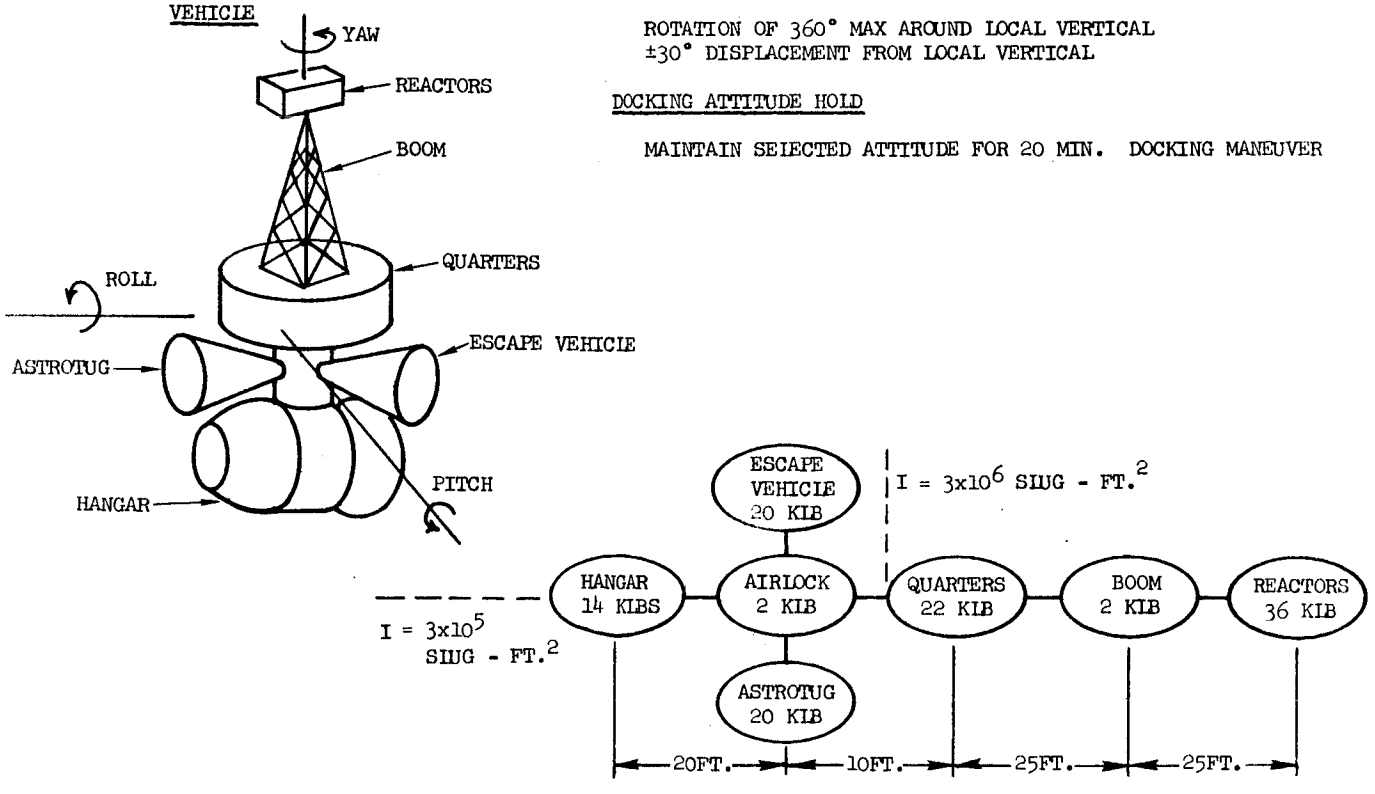



FIGURE 3-1 ORBITAL LAUNCH FACILITY REQUIREMENTS

| REQUIREMENT \ FUNCTION | RENDEZVOUS | | DOCKING | | |
|---|---|---|---|---|---|
| | PITCH YAW | ROLL | PITCH YAW | ROLL | |
| **ATTITUDE CONTROL** | | | | | |
| 1. POSITIONING ACCURACY | ±.5° | ±.5° | ±.5° | ±.5° | DEG. |
| 2. MAX. ACCELERATION | .15 | .30 | .03 | .03 | DEG./SEC.$^2$ |
| 3. THRESHOLD RATE | .05 | .05 | .05 | .05 | DEG./SEC. |
| 4. STEADY DISTURBANCES | - | - | - | - | |
| 5. VARIABLE DISTURBANCES | - | - | - | - | FT.-LBS. |
| 6. MAX. DEMAND, STEADY | $4x10^3$ | 800 | 800 | 80 | FT.-LBS. |
| 7. DURATION MAX. DEMAND | 24 | 24 | - | - | SEC. (REPEATED) |
| 8. THRESHOLD IMPULSE | $1.3x10^3$ | $1.3x10^2$ | $1.3x10^3$ | $1.3x10^2$ | FT.-LB.-SEC. |
| 9. MINIMUM PULSE-WIDTH | .5 | .5 | .5 | .5 | SEC. |
| **THRUST CONTROL** | | | | | |
| 10. MAX. FORE/AFT. ΔV | 900 | | 1.0 | | FT./SEC. |
| 11. MAX. LATERAL ΔV | - | | 1.0 | | FT./SEC. |
| 12. THRESHOLD ΔV | 1.0 | | .01 | | FT./SEC. |
| 13. MAX. DEMAND, STEADY | 4000 | | 25 | | LBS. |
| 14. DURATION MAX. DEMAND | 560 | | 100 | | SEC. (REPEATED) |
| 15. THRESHOLD IMPULSE | $2.5x10^3$ | | 25 | | LB.-SEC. |

**ORBIT TRANSFER**

REQUIREMENTS NOT ESTABLISHED

**RENDEZVOUS**

INITIAL RANGE 50 MILES MAX. (NOMINAL 10 MILES AT ALTITUDE, 3 MILES LATERALLY, 3 MILES ALTITUDE ERROR)
INITIAL VELOCITY 900 FT./SEC. MAX. (NOMINAL 450 FT./SEC.)
MANEUVER TIME 80 MINUTES.

**DOCKING**

INITIAL RANGE 500FT.
INITIAL VELOCITY ±1 FPS.
MANEUVER TIME 20 MINUTES

**VEHICLE**

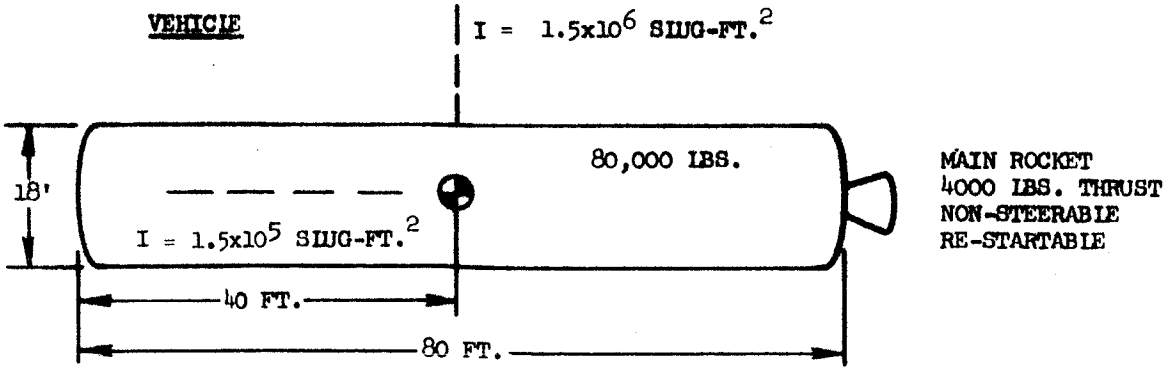



FIGURE 3-2 UNMANNED PAYLOAD REQUIREMENTS NO TUG

For example, if the torque output is M and the minimum pulse width is $T_{min}$, then

$MT_{min} \leq$ threshold impulse specification

$T_{min} \leq$ minimum pulse width specification

The specified threshold impulse will deliver the specified threshold rate, while the specified minimum pulse width assures a modulation frequency high enough to have a negligible effect on the dynamic response of the vehicle. Given a value of $T_{min}$ satisfying the second inequality, the first inequality establishes an upper limit on the maximum control torque. A limited trade-off between maximum torque and duration of maximum demand is still possible. It should be noted that the maximum torque output will be required for maximum duration a number of times during the flight.

b. Unmanned Payload, with no tug

The attitude control requirements for the unmanned payload are summarized in figure 3-2 for the no-tug situation. As shown in the figure, the unmanned payload was assumed to consist of an 80,000 lb. cylinder with a non-steerable, re-startable main propulsion unit.

When the vehicle is placed in orbit it may be a considerable distance from the desired rendezvous point and transfer orbits will probably be used to reduce the initial separation. It was not possible to establish attitude control requirements for this phase of flight because neither the initial conditions nor the guidance technique have been defined.

After reaching a point within 50 miles of the permanent facility, the rendezvous flight mode is initiated. Docking maneuvers begin at a range of 500 feet. For simplicity, the astro-tug rendezvous technique was also assumed for the unmanned payload. The details of this scheme are discussed in paragraph d. below.

Most of the attitude control system requirements tabulated in figure 3-2 were obtained in a manner similar to those for the OLF. The exceptions are thrust level and angular acceleration requirements, which are the same as those specified for the astro-tug payload combination.

c. Space Vehicle, with no tug

Requirements for the space vehicle will be similar in many respects to those of the unmanned payload. Detailed discussion is reserved for paragraph e. below.

d. Astro-tug

A sketch of the astro-tug is shown in figure 3-3 along with a table of attitude control requirements for payload retrieval. The tug must rendezvous with the incoming payload, couple itself to the payload, return to the permanent facility, and dock the payload.

The rendezvous technique selected for purposes of the attitude control system investigation was the simplest possible, and is based on the results of NASA simulator studies. A collision course is established with the objective by reducing the inertial rate of rotation of the line of sight to zero. Braking thrust is then applied along the line of sight to maintain a pre-computed schedule of range rate vs range.

| REQUIREMENT \ FUNCTION | PAYLOAD RENDEZVOUS | | PAYLOAD COUPLING | | PAYLOAD RETURN (NO HELP FROM PAYLOAD) | | PAYLOAD RETURN (HELP FROM PAYLOAD) | | PAYLOAD DOCKING (NO HELP FROM PAYLOAD) | | PAYLOAD DOCKING (HELP FROM PAYLOAD) | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | PITCH YAW | ROLL | PITCH YAW | ROLL | PITCH YAW | ROLL | PITCH YAW | ROLL | PITCH YAW | ROLL | PITCH YAW | ROLL | |
| ATTITUDE CONTROL | | | | | | | | | | | | | |
| 1. POSITIONING ACCURACY | ±.5 | ±.5 | ±.5 | ±.5 | - | - | | | ±.5 | ±.5 | | | DEG |
| 2. MAX. ACCELERATION | 2.5 | 4.5 | .03 | .03 | .15 | .3 | | | .03 | .03 | | | °/SEC$^2$ |
| 3. THRESHOLD RATE | .05 | .05 | .05 | .05 | .05 | .05 | BELOW | | .05 | .05 | BELOW | | °/SEC |
| 4. STEADY DISTURBANCES | - | - | - | - | - | - | | | - | - | | | |
| 5. VARIABLE DISTURBANCES | - | - | - | - | - | - | | | - | - | | | |
| 6. MAX. DEMAND, STEADY | 2200 | 400 | 30 | 3 | 4000 | 800 | | | 800 | 80 | | | FT. LBS |
| 7. DURATION, MAX. DEMAND | 6 | 6 | - | - | 24 | 24 | | | - | - | | | SEC. (REPEATED) |
| 8. THRESHOLD IMPULSE | 45 | 4.5 | 45 | 4.5 | 1.3x10$^3$ | 1.3x10$^2$ | NOTE | | 1.3x10$^3$ | 1.3x10$^2$ | NOTE | | FT-LB-SEC |
| 9. MINIMUM PULSE-WIDTH | 0.5 | 0.5 | 0.5 | 0.5 | 0.5 | 0.5 | | | 0.5 | 0.5 | | | SEC |
| THRUST CONTROL | | | | | | | | | | | | | |
| 10. MAX. FORE/AFT Δ V | 900 | | 1 | | 900 | | | | 1 | | | | FT/SEC. |
| 11. MAX. LATERAL Δ V | - | | 1 | | - | | | | 1 | | | | FT/SEC. |
| 12. THRESHOLD Δ V | 1 | | .01 | | 1 | | SEE | | .01 | | SEE | | FT/SEC. |
| 13. MAX. DEMAND, STEADY | 4000 | | 60 | | 4000 | | | | 25 | | | | LBS |
| 14. DURATION | 140 | | 10 | | 550 | | | | 100 | | | | SEC. (REPEATED) |
| 15. THRESHOLD IMPULSE | 600 | | 6 | | 2.5x10$^3$ | | | | 25 | | | | LB-SEC. |

PAYLOAD RENDEZVOUS

TRAVEL TO PAYLOAD; 50 MILES MAX. RANGE (NOMINAL 10 MILES AT ALTITUDE, 3 MILES LATERALLY, 3 MILES ALTITUDE ERROR)
INITIAL VELOCITY 900 FT./SEC. MAX. (450 FT./SEC. NOMINAL)
MANEUVER TIME 20 MINUTES

PAYLOAD COUPLING

INITIAL RANGE 500 FT.; INITIAL VELOCITY 1 FT./SEC.; TIME 20 MIN.

PAYLOAD RETURN

AS FOR PAYLOAD RENDEZVOUS, EXCEPT MANEUVER TIME 80 MIN.

PAYLOAD DOCKING

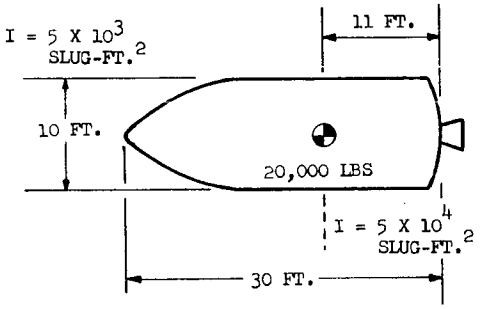
AS FOR PAYLOAD COUPLING

NOTE: MASS AND MOMENT OF INERTIA OF TUG NEGLECTED WITH RESPECT TO PAYLOAD

NOTE: PAYLOAD MAY AUGMENT TUG IN ATTITUDE AND/OR THRUST CONTROL; IF SO TUG WILL NOT PROVIDE ALL CONTROL THRUST AND/OR CONTROL TORQUES

NOTE: FOR INITIAL RENDEZVOUS WITH OLF SEE "MANNED PAYLOAD REQUIREMENTS"

NOTE: FOR OLV POSITIONING, SEE "OLV REQUIREMENTS"




MAIN ROCKET
TWO THRUST LEVELS
(4000 AND 2000 LBS.)
NON-STEERABLE
RE-STARTABLE

FIGURE 3-3 ASTROTUG REQUIREMENTS FOR PAYLOAD RETRIEVAL

The vehicle is first oriented normal to both the line of sight and the inertial angular velocity vector, after which the main propulsion unit is fired to reduce the angular velocity of the line of sight to zero. The vehicle is then re-oriented parallel to the line of sight and thrusting initiated until the range rate/range schedule is satisfied. These two operations are repeated sequentially throughout the rendezvous maneuver.

The tabulated thrust levels, angular accelerations, and rendezvous maneuver time for the tug alone were extracted from the NASA studies mentioned earlier, as were the initial ranges and relative velocities. It would appear that the initial relative velocity corresponds to an orbital plane misalignment of roughly 2°. This is quite realistic.

The maneuver time for the tug-payload combination was arbitrarily increased over that for the tug alone by an amount proportional to the payload to tug weight ratio. This was done to keep control forces at reasonable levels. Note that the total trip time for the tug is above two hours. Further time extensions might cause serious operator fatigue.

The attitude angle accuracy, velocity threshold, and angular rate threshold specified for the docking maneuver were obtained from documents supplied by Vought Astronautics. These represent the conditions required at the end of the docking maneuver. Similarly, the rendezvous phase specifications correspond to the conditions required by Vought at the initiation of the docking maneuver.

e. Unmanned Payload, with Tug

The attitude control requirements for the unmanned payload are summarized in figure 3-4 for the case when a tug is available. In this case, the payload attitude control system is required to damp the initial tumbling and, upon command from the tug, hold steady while the tug attaches itself.

Depending upon the capability of the tug, the payload may have to provide all or part of the torque and/or thrust required for the return trip and for docking. The question of whether or not to couple the payload reaction jets into the tug control system after the vehicles are joined has not been resolved.

f. Orbit-Launched Vehicle, with Tug

A possible Orbit-Launched Vehicle (OLV) configuration is shown in figure 3-5. It is constructed of a modified Saturn S-IV, two Agena B's, and an Apollo capsule. Prior to orbital launch, an attitude control system is required to stabilize the vehicle in order to avoid hitting the permanent facility, to control the vehicle while it taxis to the launch site, and to orient the vehicle for launch. As with the unmanned payload, the OLV attitude control system requirements vary according to whether or not a tug is available and if available, whether or not the OLV reaction jets are coupled into the tug control system. All three possibilities are included in figure 3-5.

Taxi to the launch site was considered to begin with an initial small velocity impulse followed by a long coasting phase. The vehicle will be re-oriented while

| REQUIREMENT \ FUNCTION | INITIAL ALIGNMENT | | COUPLING ATTITUDE HOLD | | TUG AUGMENTATION ATTITUDE CONTROL | | TUG AUGMENTATION THRUST CONTROL | | |
|---|---|---|---|---|---|---|---|---|---|
| ATTITUDE CONTROL | PITCH YAW | ROLL | PITCH YAW | ROLL | PITCH YAW | ROLL | PITCH YAW | ROLL | |
| 1. POSITIONING ACCURACY | ±5° | ±5° | ±.5 | ±.5 | - | - | — | — | DEG. |
| 2. MAX. ACCELERATION | $10^{-3}$ | $10^{-2}$ | - | - | .15 | .3 | — | — | DEG./SEC.$^2$ |
| 3. THRESHOLD RATE | .05 | .05 | .05 | .05 | .05 | .05 | — | — | DEG./SEC. |
| 4. STEADY DISTURBANCES | 2.5 | - | 2.5 | - | - | - | — | — | FT.-LBS. |
| 5. VARIABLE DISTURBANCES | .045 | - | .045 | - | - | - | — | — | FT.-LBS. |
| 6. MAX. DEMAND, STEADY | 26 | 26 | - | - | 4000 | 800 | — | — | FT.-LBS. |
| 7. DURATION MAX. DEMAND | 270 | 540 | - | - | 24 | 24 | — | — | SEC. (REPEATED) |
| 8. THRESHOLD IMPULSE | $1.3\times10^3$ | $1.3\times10^2$ | $1.3\times10^3$ | $1.3\times10^2$ | $1.3\times10^3$ | $1.3\times10^2$ | — | — | FT.-LBS.-SEC. |
| 9. MINIMUM PULSE-WIDTH | 0.5 | 0.5 | 0.5 | 0.5 | 0.5 | 0.5 | — | — | SEC |
| THRUST CONTROL | | | | | | | | | |
| 10. MAX. FORE/AFT $\Delta V$ | — | | — | | — | | 900 | | FT./SEC. |
| 11. MAX. LATERAL $\Delta V$ | — | | — | | — | | 1 | | FT./SEC. |
| 12. THRESHOLD $\Delta V$ | — | | — | | — | | .01 | | FT./SEC. |
| 13. MAX. DEMAND, STEADY | — | | — | | — | | 4000 | | LBS. |
| 14. DURATION MAX. DEMAND | — | | — | | — | | 550 | | SEC. (REPEATED) |
| 15. THRESHOLD IMPULSE | — | | — | | — | | 25 | | LB.-SEC. |

INITIAL ALIGNMENT

DAMPING 1 RPM INITIAL ROTATION IN ROLL
MANEUVER TIME 90 MINUTES

COUPLING ATTITUDE HOLD

HOLD SELECTED ATTITUDE FOR 20 MINUTE MANEUVER

TUG AUGMENTATION (ATTITUDE CONTROL)

ACCEPT COMMANDS FROM TUG AND PROVIDE CONTROL TORQUES

TUG AUGMENTATION (THRUST CONTROL)

ACCEPT COMMANDS FROM TUG AND PRIVIDE THRUST

NOTE:
TUG DESIGN WILL SPECIFY WHICH AUGMENTING FUNCTIONS ARE REQUIRED, IF ANY

NOTE:
SEE "UNMANNED PAYLOAD REQUIREMENTS" FOR RENDEZVOUS - DOCKING BREAKDOWN

VEHICLE:

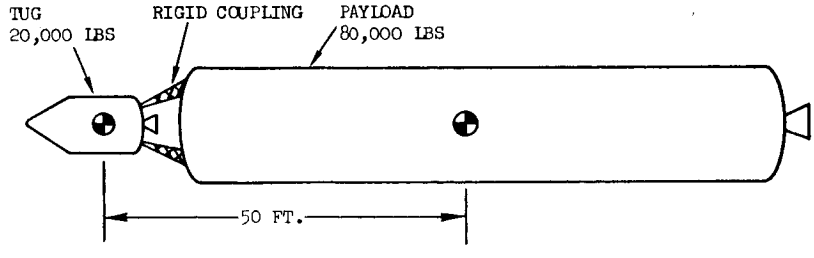



FIGURE 3-4 UNMANNED PAYLOAD REQUIREMENTS WITH TUG

| REQUIREMENT \ FUNCTION | ATTITUDE HOLD NEAR OLF | | TAXI TO LAUNCH SITE (NO TUG) | | LAUNCH ATTITUDE AND POSITION HOLD | | |
|---|---|---|---|---|---|---|---|
| ATTITUDE CONTROL | PITCH YAW | ROLL | PITCH YAW | ROLL | PITCH YAW | ROLL | |
| 1. POSITIONING ACCURACY | ±2° | ±2° | ±5° | ±5° | ±2° | ±2° | DEG. |
| 2. MAX. ACCELERATION | - | - | $2x10^{-4}$ | $2x10^{-4}$ | $2x10^{-4}$ | $2x10^{-4}$ | DEG./SEC.$^2$ |
| 3. THRESHOLD RATE | .02 | .02 | .05 | .05 | .02 | .02 | DEG./SEC. |
| 4. STEADY DISTURBANCES | .33 | - | - | - | .33 | - | FT.-LB. |
| 5. VARIABLE DISTURBANCES | - | - | - | - | - | - | |
| 6. MAX. DEMAND, STEADY | - | - | 20 | 2 | 20 | 2 | FT.-LBS. |
| 7. DURATION MAX. DEMAND | - | - | 900 | 900 | 900 | 900 | SEC. (REPEATED) |
| 8. THRESHOLD IMPULSE | $2x10^3$ | $2x10^2$ | $5x10^3$ | $5x10^2$ | $2x10^3$ | $2x10^2$ | FT-LB-SEC. |
| 9. MINIMUM PULSE-WIDTH | 0.5 | 0.5 | 0.5 | 0.5 | 0.5 | 0.5 | SEC |
| THRUST CONTROL | | | | | | | |
| 10. MAX. FORE/AFT. ΔV | — | | 4.4 | | 1.0 | | FT./SEC. |
| 11. MAX. LATERAL ΔV | — | | 1.0 | | 1.0 | | FT./SEC. |
| 12. THRESHOLD ΔV | — | | 1.0 | | 1.0 | | FT./SEC. |
| 13. MAX. DEMAND, IMPULSE | — | | $2x10^4$ | | $4.7x10^3$ | | LB.-SEC. |
| 14. DURATION MAX. DEMAND | — | | - | | - | | |
| 15. THRESHOLD IMPULSE | — | | $4.7x10^3$ | | $4.7x10^3$ | | LB.-SEC. |

ATTITUDE HOLD NEAR OLF

SLAVE TO OLF ATTITUDE (LOCAL VERTICAL)

TAXI TO LAUNCH SITE

TAXI 3 MILES IN ONE HOUR; ROTATE 180° FOR RETROTHRUST

LAUNCH ATTITUDE AND POSITION HOLD

ROTATE 180° MAX. FOR LAUNCH; HOLD ATTITUDE AND POSITION AS COMMANDED FOR OLF

NOTE:

IF TUG IS USED WITHOUT TUG/OLV CONTROL SYSTEM COUPLING THE OLV HAS NO TAXI REQUIREMENTS.

IF OLV ATTITUDE CONTROL SYSTEM IS CONTROLLED FROM TUG THE ATTITUDE CONTROL FUNCTIONS LISTED ABOVE STILL APPLY BUT THE TUG WILL PROVIDE THRUST FOR TAXI.

THE TUG CANNOT ASSIST THE OLV IN LAUNCH ATTITUDE AND POSITION HOLD.

NOTE:

INJECTION, MIDCOURSE AND LUNAR ATTITUDE AND THRUST VECTOR CONTROL PROVIDED BY SATURN S-IV, AGENDA-B, AND APOLLO CONTROL SYSTEM.

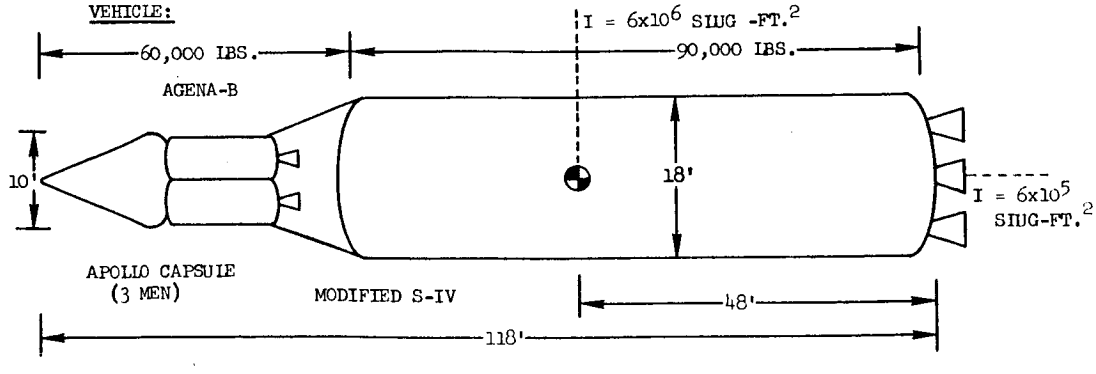



FIGURE 3-5 SPACE VEHICLE REQUIREMENTS

coasting so that a retro-thrust can be applied when the launch site is reached. The OLV will then be slewed to the proper launch direction as computed and commanded by the OLF.

It does not appear feasible to launch directly from the permanent facility because of the danger to the facility crew and because both vehicles are of roughly the same size. Any impacts may seriously perturb the OLF orbit (and structure). For similar reasons, the tug cannot assist the OLV in the final phases of pre-launch attitude hold.

The Saturn S-IV is presently expected to have a restart capability. In order to provide this capability, a coast attitude control system will be added so that adequate attitude control will be available when the main engines are off. Such a system has been sized by NASA, utilizing gaseous $H_2O_2$ as the propellant and swiveling reaction nozzles as torque generators. If available, this Saturn modification may well provide sufficient prelaunch attitude control and make an OLO-supplied piggyback unit unnecessary.

## 3-2. COMPARISON OF ALTERNATE ORBITAL LAUNCH OPERATION CONFIGURATIONS

The information assembled in the preceding section can be used to develop a comparison of the permanent-orbiting-facility/no-permanent-facility and tug/no-tug complexes mentioned earlier. The results of these comparisons, discussed further below, can be summarized very briefly:

any attitude-control equipment trade-offs seem relatively unimportant when compared to the major operational variations implied by the alternate configurations.

The major difference between the permanent-facility and no-permanent-facility configurations from the standpoint of attitude control appears to be one of equipment duplication in successive operations. If a permanent facility is available, any special equipment required for rendezvous can be reused. If no permanent facility is available, this equipment will presumably be ejected as waste before the space vehicle is injected into its cislunar trajectory. The ejected equipment will be lost and booster fuel will have to be expended in replacing it for later operations. It does appear, incidentally, that special attitude sensors will be required for docking.

Otherwise, there appears to be no major difference between the permanent-facility and no-permanent-facility configurations from the standpoint of attitude control. The former requires another vehicle and thus another attitude-control system, and some minor changes may be made in the vehicle attitude-control systems if a permanent facility is available. However, the most pressing OLO attitude-control problem, the control of the relative attitude of two vehicles in close proximity to one another, must be solved in either case.

Attitude-control requirements are also somewhat sensitive to the presence or absence of an astrotug. For example, rough calculations of the fuel required for torque and thrust, with and without a tug, lead to the tabulation of table 3-1.

TABLE 3-1

FUEL REQUIREMENTS

| Phase of Payload Retrieval Operation | Thrust Requirement | Lbs fuel req'd for thrust | | Attitude Control Required | Lbs Fuel req'd for torque ($I_{sp}$ = 200) | | |
|---|---|---|---|---|---|---|---|
| | | Without Tug | With Tug | | Without Tug | With Tug Coupled | Not coupled |
| 1. Rendezvous with Payload | $\Delta V$ = 900FPS ($I_{sp}$ = 300) | 0 | 1870 | 3 max. demand max. duration maneuvers | 0 | 20 | 20 |
| 2. Coupling to Payload | 3 Max. demand, max. duration maneuvers ($I_{sp}$ = 300) | 0 | 15 | 3 max. demand, max. duration maneuvers | 0 | negligible | negligible |
| 3. Payload Return | $\Delta V$ = 900FPS ($I_{sp}$ = 300) | 7,330 | 9200 | 3 max. demand, max. duration maneuvers | 140 | 280 | 280 |
| 4. Payload Docking | 3 max. demand, max. duration maneuvers ($I_{sp}$ = 200) | 38 | 45 | 3 max. demand, max. duration meneuvers | negligible | negligible | 93 |
| 5. Totals (20,000 lb tug) | - | 7368 | 11,130 | - | 140 | 300 | 393 |

The rather striking differences in fuel requirements are due to the following factors:

The 20,000-pound tug must be taken to the payload and returned to the facility

The tug, positioned 50 feet off the tug/payload CG, increases the vehicle mass by 25 percent and doubles the moment of inertia

If the tug cannot operate the payload reaction jets (i.e., no electrical coupling) then a lateral vernier thrust applied by the tug must be accompanied by a large compensating torque to maintain a fixed attitude during docking maneuvers.

Since the weight of the tug is not firmly established, totals have also been shown for a 10,000 pound vehicle. The use of either tug for the entire rendezvous maneuver involves a severe fuel-weight penalty. If use of the tug is restricted to the docking maneuver, the weight penalties are considerably reduced, as shown below.

Total Docking Fuel Weights

| | |
|---|---|
| No tug: | 38 pounds |
| 10,000-pound tug: | 75 pounds if coupled electrically, 230 pounds if not. |
| 20,000-pound tug: | 60 pounds if coupled electrically, 153 pounds if not. |

If one assumes that the tug is a dual-purpose vehicle which would be in orbit whether used during the docking operation or not, then the use of the tug is justified if 13-27 pounds of equipment can be eliminated in the incoming payload, unless electrical interconnection of tug and payload is impractical. If the tug must operate independently, then 230-244 pounds of equipment must be eliminated to justify its use. Thus, if the tug and payload can be interconnected electrically, use of a tug appears promising. If not, the tug seems somewhat marginal.

It must be emphasized that operational advantages gained with the tug could well outweigh the weight penalties discussed above. Such factors, which are beyond the scope of an attitude-control system study, have not been discussed in this document. Finally, it should be mentioned that attitude control of all vehicles in any of the OLO configurations is well within the state-of-the-art.

## 3-3. VEHICLE ATTITUDE CONTROL SYSTEM SUMMARY

A summary of the detailed discussion to be given in Section IV is presented below. The major elements of the attitude-control systems recommended for the orbital launch facility, the unmanned payload, and the astrotug are presented. Tabulations of power, weight, and volume requirements are included.

The attitude-control system recommended for the orbital launch facility is quite conventional. A horizon seeker serves as the pitch-and-roll reference, slaving the vehicle to the local vertical. An integrating rate gyro is used to stabilize the vehicle in yaw. Periodic manual

realignment of the yaw axis will probably be required, although automatic alignment is feasible. A capability for manual reorientation prior to docking is provided, along with a fine attitude-hold mode for use during docking maneuvers. Torques are obtained from a cold-nitrogen reaction-jet control system. A storable bi-propellant system would offer a slight weight advantage, but is presently less reliable than the cold-nitrogen system. Table 3-2 contains summary information pertaining to the OLF attitude control system.

The attitude of the unmanned payload is controlled remotely from the Orbiting Facility. An inertial reference unit is required in the payload to derive vehicle attitude in the Facility co-ordinate system. The most significant attitude control requirement for the unmanned payload is the wide range of torques necessary to perform the requisite maneuvers. Orientation of the vehicle for thrusting during rendezvous requires large torques, while the docking procedure will require very small torques. Although a cold nitrogen system is recommended because of its higher reliability, the high torque required for rendezvous places a definite weight advantage with the storable bi-propellant system. Summary data is presented in table 3-3.

The components required for attitude control of the manual Astrotug are listed in table 3-4. The system in functionally quite similar to the unmanned payload, except that control is exercised directly rather than remotely. The recommended method of torque generation in this case is by utilization of boil-off hydrogen from the main propulsion tanks.

Attitude control requirements for the unmanned payload are listed in table 3-5 for the case when a tug is available. Comparing with table 3-3, it can be seen that the presence of a tug eliminates considerable weight from the payload itself. However, as discussed in the preceding section, the overall fuel weight requirements (tug plus payload) will be higher than for the payload alone.

TABLE 3-2

ORBITAL LAUNCH FACILITY
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

| Component | Weight (lbs.) | Volume ($ft.^3$) | Power (V.A.) | Remarks |
|---|---|---|---|---|
| 1 Integrating Rate Gyro | 2 | 0.02 | 13 | |
| 1 Three Axis Rate Gyro Assembly | 3 | 0.04 | 24 | |
| 1 Panel Controller | 3 | 0.05 | 1 | |
| 1 Electronic Assembly | | | | |
| 3 Threshold Detector Amplifiers | 15 | 0.25 | 50 | All D.C. power required is self generated |
| 1 Gyro Torquer Amplifier | | | | |
| 3 Guidance Couplers Power & Switching Components | | | | |
| Cold Nitrogen Reaction Jet Control System; Nozzles, Valves, etc. | 55 | - | - | Alternate system using storable bipropellant; Total wt. = 69.5 lbs |
| Propellant Weight for 30 Days | 24.5 | | | |
| Totals | 102.5 | 0.31 | 88 | With alternate reaction control system Total wt. = 92.5 lbs |
| ASSOCIATED GUIDANCE EQUIPMENT | | | | |
| 2 Horizon Seekers | | | | |
| 1 Long Term Yaw Attitude Reference | | | | |
| 1 Telemetry Receiver | | | | Provides capability for remote initiation of docking hold mode for initial rendezvous |

TABLE 3-3

UNMANNED PAYLOAD (NO TUG)
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

| Component | Weight (lbs.) | Volume ($ft.^3$) | Power (V.A.) | Remarks |
|---|---|---|---|---|
| 3 Integrating Rate Gyros | 6 | .06 | 39 | |
| 4 Linear Accelerometers | 3 | .03 | 6 | |
| 1 Inertial Attitude Reference Unit | 50 | 1.5 | 60 | |
| 1 Electronic Assembly | | | | |
| 3 Threshold Detector Amplifiers | | | | |
| 3 Gyro Torquer Amplifiers Power & Switching Components | 20 | .4 | 66 | All D.C. power required is self generated |
| 2 Electromechanical Integrators | | | | |
| Cold Nitrogen Attitude and Vernier Velocity Reaction Jet Control; Nozzles, Valves, etc. | 263 | - | - | Alternate system using storable bi-propellant |
| Propellant Weight for Rendezvous & Docking | 140 | | | Total wt. = 152 |
| Totals | 479 | 2 | 171 | With alternate reaction control system, Total wt. = 228 |
| ASSOCIATED GUIDANCE EQUIPMENT | | | | |
| 1 Telemetry Transmitter-Receiver | | | | |
| 1 Radar Transponder | | | | |

TABLE 3-4
MANNED ASTROTUG (MINIMUM SYSTEM)
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

| Component | Weight (lbs.) | Volume (ft.3) | Power (V.A.) | Remarks |
|---|---|---|---|---|
| 1 3 Axis Rate Gyro Assembly | 3 | 0.04 | 24 | Cannot be specified until optimum division of attitude control functions between tug and payload is established. Quoted weight corresponds to electrically coupled tug-payload when tug is used for rendezvous and docking |
| 1 Panel Controller | 3 | 0.05 | 1 | |
| 1 Electronic Assembly | | | | |
| 3 Threshold Detector Amplifiers | 10 | 0.20 | 33 | |
| Power and Switching Components | | | | |
| Hydrogen Boil-Off Attitude and Vernier Velocity Reaction Jet Controls; Nozzles, Valves, etc. | 113 | — | — | |
| Propellant Weight | 160 | | | |
| Totals | 289 | 0.29 | 58 | |
| ADDITIONAL EQUIPMENT REQUIRED FOR AUGMENTED SYSTEM | | | | |
| 1 Inertial Attitude Reference Unit | 50 | 1.5 | 60 | |
| 1 Resolving Computer | 10 | 0.20 | 25 | |
| 1 Electronic Assembly | | | | |
| 5 Electromechanical Integrators | 6 | 0.15 | 25 | |
| Additional Power and Switching Components | | | | |
| 4 Linear Accelerometers | 3 | 0.03 | 6 | |
| ASSOCIATED GUIDANCE EQUIPMENT | | | | |
| Tracking Radar Gimbal Pickoffs | | | | |

TABLE 3-5

UNMANNED PAYLOAD (WITH TUG)
COMPONENT WEIGHT, VOLUME, AND POWER ESTIMATES

| Component | Weight (lbs.) | Volume ($ft.^3$) | Power (V.A.) | Remarks |
|---|---|---|---|---|
| 3 Integrating Rate Gyros | 6 | .06 | 39 | |
| 1 Electronic Assembly<br>3 Threshold Detector Amplifiers<br>3 Gyro Torquer Amplifiers | 15 | .25 | 50 | |
| Power and Switching Components | | | | |
| Attitude Reaction Jet Controls; Nozzles, Valves, etc. | 263 | - | - | Cannot be specified until optimum division of attitude control functions between tug and payload is established. Quoted weight corresponds to electrically coupled tug-payload when tug is used for rendezvous and docking. |
| Propellant Weight | 140 | | | |
| Total | 424 | .31 | .89 | |

The realistic determination of the reliability of a complex attitude control system is an extremely difficult task. Reliability estimates were not included in the summary tables because reliability depends so heavily on factors which have not yet been specified. Examples of such factors include the packaging design details, type of circuitry employed, environment, component limitations, degree of redundancy, and maintenance considerations. However, quantitative reliability goals compatible with the present state-of-the-art have been established for the OLO attitude control equipment. These goals, the basis for their establishment, and the program required to reach them are discussed in the following paragraphs.

The first step in establishing a reliability goal is to estimate the capabilities of state-of-the-art equipment. Failure-rate data for a typical single-axis, attitude-control system is tabulated and summed in table 3-6.

The pressure regulator, valving, tank, and chamber failure rates have been extrapolated from Sperry missile experience. Applying the usual reliability formula

$$R = e^{-kt}$$

leads to the reliabilities listed in table 3-7.

TABLE 3-6

COMPONENT FAILURE RATES

| Component | Failure Rate, K, failures per million hours |
|---|---|
| Threshold Detector Amplifier | 127 |
| Gyro Torquer Amplifier | 36 |
| Coupler | 72 |
| Rate Gyro | 400 |
| Integrating Rate Gyro | 600 |
| Pressure Regulator | 120 |
| Valves | 50 |
| Tanks | 5 |
| Chamber | 15 |
| Sub-total | 1425 |
| Miscellany | 275 |
| Single-Axis Attitude Control System Failure Rate | 1700 |

TABLE 3-7

SINGLE-AXIS ATTITUDE CONTROL SYSTEM RELIABILITY

| Time of mission | Probability of completing mission without a failure in one axis of attitude control system. |
|---|---|
| one day | 0.96 |
| one week | 0.75 |
| one month | 0.32 |

Most of the vehicles will be manned so that maintenance will be possible, and the mission duration of the unmanned vehicles will be much shorter than shown above. **In any case, three important conclusions can be drawn:**

- The reliability of OLO attitude control equipment must be better than the state of the art.
- Periodic maintenance and/or replacement must be possible. This, in turn, means that modular construction must be utilized and that emergency operating modes must be provided for use during maintenance.
- Redundant circuitry must be provided to reduce the failure rate by 50 percent. This can be accomplished by providing automatic or semi-automatic replacement of some of the more critical components, such as the rate gyros.

Based on the above calculations, a failure-rate goal of less than 500 failures per million hours for each axis of the attitude-control system in a typical OLO vehicle must be obtained. This will increase reliability to the order of 0.99 for a one-day period. This goal represents a definite improvement over the state-of-the-art, yet is both realistic and attainable.

# SECTION IV

# RECOMMENDED SYSTEMS AND TRADE-OFF CONSIDERATIONS

## 4-1. ORBITAL LAUNCH FACILITY

### a. Recommended System

#### (1) System Description

This section describes an attitude stabilization and control system for the OLF vehicle. The control system, completely contained in the OLF, starts to function at booster separation. The vehicle is aligned and controlled to the local vertical. Manual control and remote guidance input are provided to allow vehicle reorientation for rendezvous, payload docking and probe launching.

The following description is presented by control axis (pitch, roll and yaw) broken down into operating modes. The system operating modes considered are those presented in Section III.

- Initial Stabilization and Alignment
- Coarse Attitude Hold
- Docking Alignment
- Fine Attitude Hold
- Emergency Manual Control

##### (a) Pitch Axis (figure 3-1)

It is assumed that the OLF's pitch axis is required to be automatically aligned to the local vertical. In addition,

the system must be capable of manual or remote attitude re-orientation. Figure 4-1 is a block diagram of the basic approach to pitch-axis stabilization and control.

## 1. Initial Stabilization and Alignment

The OLF-Booster separation will impart torques to the OLF that produce vehicle tumbling. The tumbling rates, assumed to be in the order of one rpm, must be counteracted by the stabilization and control system to align the vehicle to the control reference. For this discussion, the control reference is assumed to be the local vertical.

The control signal for initial alignment is derived from body-mounted horizon seekers. The error signal, proportional to the error between the vehicle's pitch axis and the local vertical, is filtered and fed to the ASCS threshold detector. If the attitude error exceeds five degrees, the threshold detection fires the pitch reaction jets, torquing the vehicle towards zero attitude error. The vertical attitude error is summed with a pitch-rate gyro that provides the damping necessary to control the on-off limit cycle. The need for dual-reaction jet capability and the type of threshold detector employed (i.e. pulse-width modulator or conventional on-off) will be determined as a result of proposed trade-off studies. The alignment sequence is programmed to start after the vehicle is in orbit.

## 2. Coarse Attitude Hold

The system configuration for coarse attitude hold is the same as for the initial alignment. The vehicle's pitch attitude is held to within five degrees of the local vertical. The system's angular rate threshold could be set above the

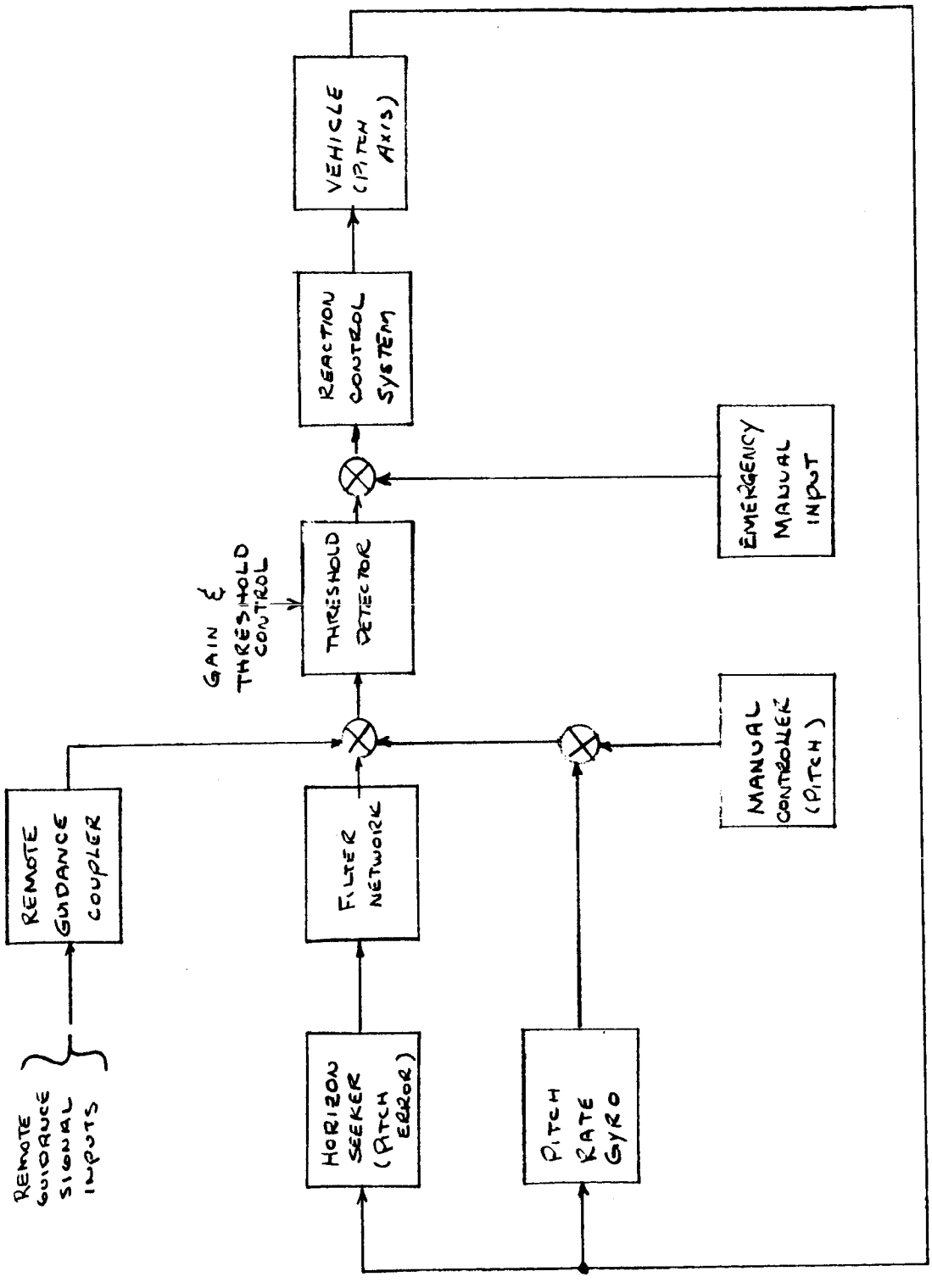



FIGURE 4-1 STABILIZATION AND CONTROL PITCH AXIS BLOCK DIAGRAM

4-degree-per-minute vehicle pitch rate required to track the local vertical, in order to prevent reaction control actuation for a satisfactorily stabilized vehicle.

During this mode of operation an integrating rate gyro could also be used for the basic attitude-stabilization reference, with the horizon seeker providing a back-up reference. This implementation would provide additional flexibility and more reliability. Utilizing an integrating rate gyro would also make the use of an electronically derived rate possible, eliminating the need for a rate gyro in this mode.

### 3. Docking Orientation and Fine Attitude Control

During rendezvous and docking of a payload with the OLF, the OLF's attitude will probably be required to be automatically held to one degree or less, and the vehicle must be capable of being reoriented, remotely or manually, to facilitate the mating of the OLF to the incoming vehicle.

By remote or local selection, the system's attitude and rate thresholds will be tightened to 0.5 degree and 0.05 degree per second, respectively, to provide a fine attitude control mode. The remote selection is required for rendezvous and docking of the initial incoming payload with the unmanned OLF. Remote guidance signals could conceivably be coupled to the OLF's reaction-control system to provide a capability for remotely controlling the OLF's pitch attitude for alignment during docking, but this does not appear absolutely necessary.

For a manned OLF, a manual controller is provided with a $\pm$ 30 degree attitude command capability. In addition to the pitch-attitude control, the controller provides fine attitude-control selection and gain control. A gain selector

is provided to compensate for large moment-of-inertia changes that can possibly occur when large payloads are accumulated on the OLF. Manual gain programming is proposed in lieu of a more complex adaptive control. Further study will be required to establish the need for gain control, as well as to confirm that the suggested approach be implemented.

4. Emergency Manual Control

An emergency manual control capability is provided. Provision is made for direct coupling of an emergency control to the reaction jet system to enable manual override for OLF attitude control. The override will be either through direct electrical inputs to the reaction control valve torquer or through mechanical override of the torquer output. The fail-safe characteristics of each will be considered prior to final selection.

The reaction jet controls will be authority limited for safety. Full-on (control hardover) will not disable the men on board the OLF. In the event of a system malfunction, emergency manual control is assumed to stabilize the vehicle and allow replacement of the malfunctioning equipment. Spare replacement equipment is proposed in lieu of redundancy in the interests of a less complex and therefore more reliable configuration.

(b) Roll Axis

The roll axis, shown in figure 4-2, is functionally the same as the previously described pitch axis.

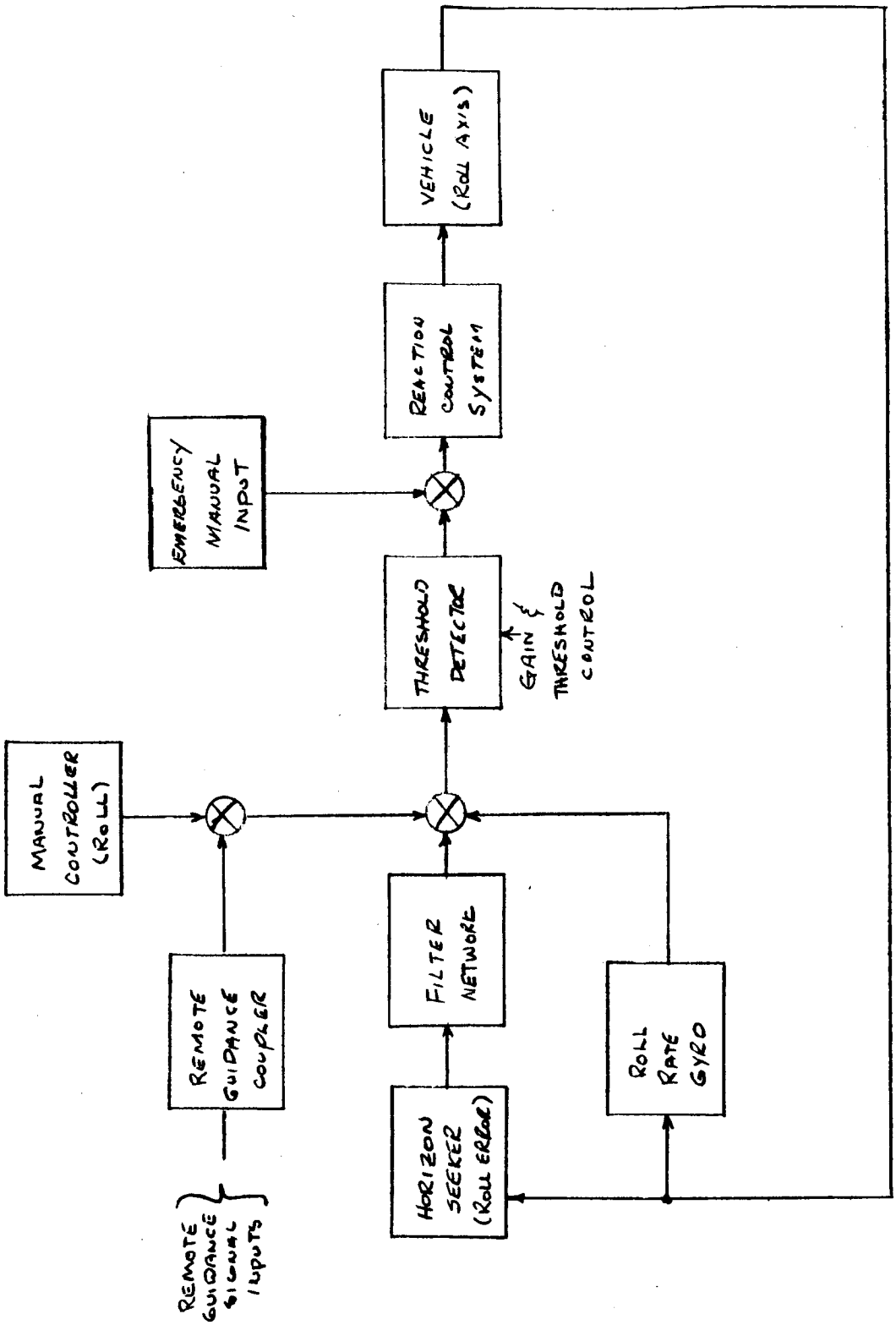



FIGURE 4-2 STABILIZATION AND CONTROL ROLL AXIS BLOCK DIAGRAM

(c) Yaw Axis

The yaw axis, shown in figure 4-3, functions in a manner similar to the roll and pitch axis except for its control reference(s) and method of manual control. The control reference for the yaw axis during initial alignment and periodic realignment phases of OLF operation is the forward velocity vector (orbit direction). Vehicle alignment to the velocity vector is desirable in order to provide proper pitch axis orientation with its attendent fuel efficiency. When the vehicle is stabilized to within the system's deadzone, the control reference is switched to a synchronized integrating rate gyro, (i.e. the gyro follows-up on itself until the switching point is reached) and the gyro then becomes the attitude reference. Deviations of the vehicle's attitude are sensed by the integrating rate gyro and when the attitude error exceeds 5 degrees (0.5 degree for fine control) or rate error exceeds the threshold, the reaction jets restore the reference attitude.

Manual or remote guidance inputs are fed through a gyro torquer amplifier to the integrating rate-gyro torque motor. This produces a gyro precession rate proportional to the input voltage. As the gyro is precessed, the vehicle attitude error grows. The yaw reaction jets are then actuated to command a yaw rate proportional to the input command. Upon removal of the command signal, (either from the manual controller or via remote guidance) the gyro stops precessing and a new reference attitude is established. Full-freedom (360 degree) yaw-attitude reorientation is available to facilitate payload rendezvous and docking. The vehicle should be manually reoriented to the velocity vector after completion of the docking maneuver.

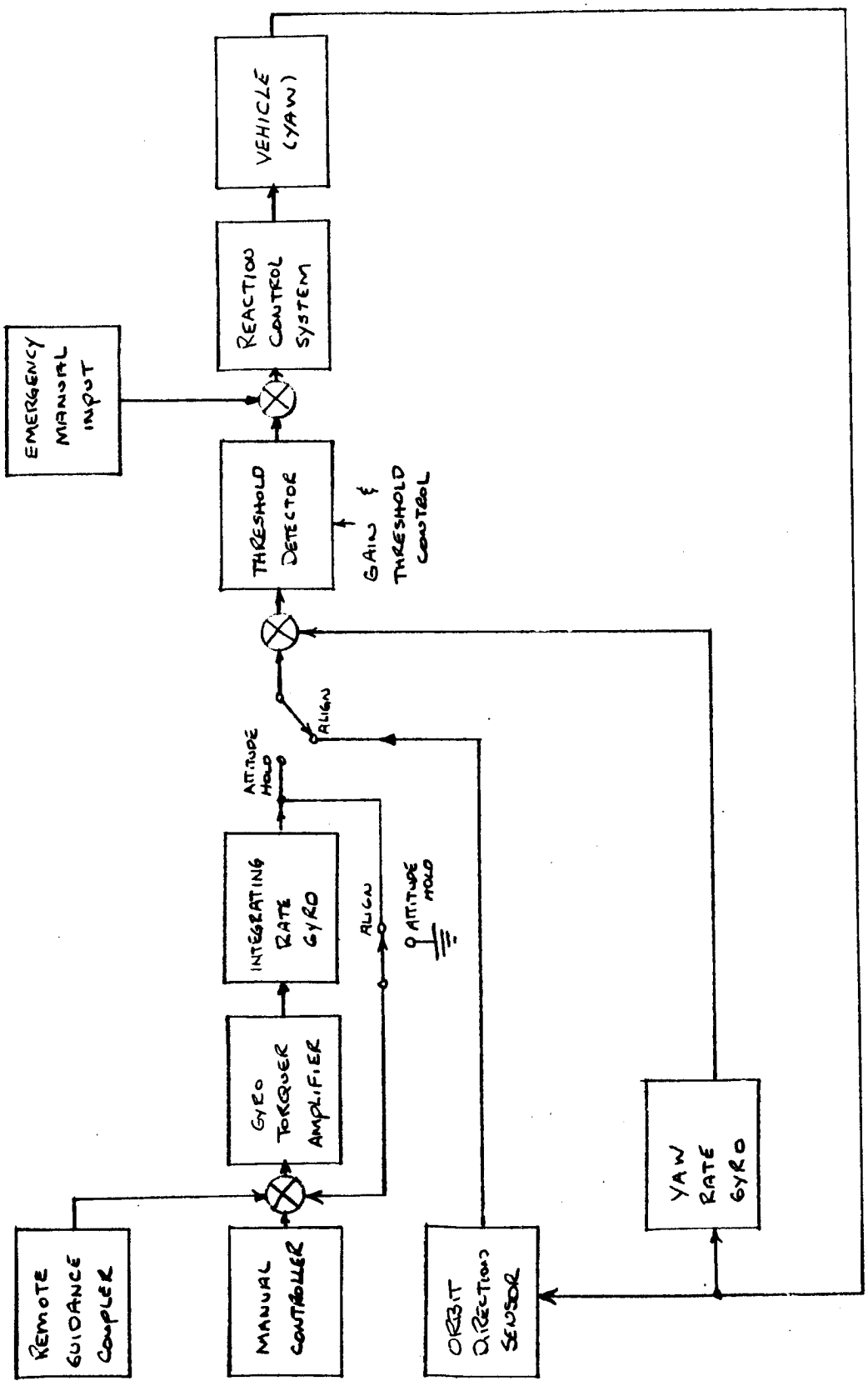



FIGURE 4-3 STABILIZATION AND CONTROL YAW AXIS BLOCK DIAGRAM

(2) Sensors

The equipments peculiar to the stabilization and control system are listed below:

| Component | Volume-$Ft^3$ | Weight-Lbs | *Power (115V,400cps)VA |
|---|---|---|---|
| Integrating Rate Gyro | 0.02 | 2 | 13 |
| Three-Axis Rate Gryo | 0.04 | 3 | 24 |
| Controller | 0.05 | 3 | 1 |
| Electronic Ass'y. | 0.2 | 15 | 50 |
| • Three threshold-detector amplifiers | | | |
| • One gyro torquer amplifier | | | |
| • Three remote-guidance couplers | | | |
| • Power and switching components | | | |

*NOTE: All d-c power required is self generated.

The equipments required for system operation that are not considered peculiar to the stabilization and control system (i.e. equipment necessary for vehicle guidance) are as follows:

| Component | Function |
|---|---|
| Two Horizon Seekers | To provide signals proportional to roll and pitch body axis deviations from the local vertical |

| Component | Function |
|---|---|
| Orbit Direction Sensor | To provide an output proportional to the vehicle's yaw axis orientation with respect to its velocity vector (orbit direction). |
| Navigation Receiver | To provide the control signals for remote control of the vehicles attitude (three axis). |

The characteristics of the sensors peculiar to the stabilization and control system are as follows:

(a) Integrating Rate Gyro Assembly

This single axis assembly will mount one miniature integrating rate gyro with its required excitation circuitry. This gyro will be sensitive to the vehicles yaw attitude. The Sperry SYG 1000 is an integrating rate gyro whose performance would satisfy the requirements of the proposed OLF stabilization and control system. Performance characteristics for this unit are as follows:

| | |
|---|---|
| Mass Unbalance (any axis) | $0.2^{\circ}$/hr/g |
| Fixed torque restraint at null | $1^{\circ}$/hr max. |
| Random Drift - Azimuth position | $0.007^{\circ}$/hr |
| Vertical position | $0.005^{\circ}$/hr |
| Anisoelastic Drift | $0.01^{\circ}$/hr/$g^2$ |
| Warm-up time | 10 min. max. |
| weight (less excitation and mounting) | 0.95 lb. |

(b) Rate Gyro Assembly

This three axis rate gyro assembly will mount three miniature rate gyroscopes with their required excitation circuitry. This unit will be sensitive to vehicle yaw, pitch and roll rates. Typical performance characteristics of a unit currently used are as follows:

| | |
|---|---|
| Threshold | $0.01^{\circ}$/sec. |
| Null | $0.03^{\circ}$/sec. |
| Full scale rate | $\pm$ $20^{\circ}$/sec. |
| Full scale output | 5.6 volts |
| Motor and Pickoff Excitation | 26 volts, 400 cps, single phase |
| Natural frequency | 14 cps |
| Damping Ratio | $0.7 \pm 0.3$ |
| Weight | 3 pounds max. |

NOTE: Both the above units are available with self contained heaters.

The stabilization and control system electronics will employ modular construction utilizing silicon semi-conductors throughout. Signal and interlock switching will be accomplished by solid state devices (no relays will be used). Circuit redundancy will be considered where significant reliability improvement can be realized. The system will utilize 115 volt, 400 cps, single phase power only. All necessary three phase AC and DC will be self generated.

(3) Torque Generators

Three types of reaction jet systems were considered for the orbital launch facility. They are: cold nitrogen, storable bi-propellant, and $H_2$-$O_2$ bi-propellant systems. An analysis of the OLF attitude control requirements for the three systems resulted in the following weight breakdown:

| Type of System | Propellant Weight per 30 day period at 0.75 lbs/day | Tare Weight | System Weight |
|---|---|---|---|
| | lbs. | lbs. | lbs. |
| Cold Nitrogen | 24.5 | 55 | 79.5 |
| Storable Bi-propellant | 24.5 | 57 | 81.5 |
| $H_2$-$O_2$ Bi-propellant | 24.5 | 45 | 69.5 |

The candidate systems were rated on the basis of certain parameters which were weighted by appropriate sensitivity factors. The selection parameters and the corresponding sensitivity factors are:

| | |
|---|---|
| Reliability | 1.0 |
| Weight | 1.0 |
| Volume | 0.9 |
| Power | 0.9 |
| Availability | 0.8 |
| State-of-Art development required | 0.8 |
| Maintainability | 0.8 |

As a result of this trade-off procedure, the cold-nitrogen system was selected as the recommended system. This system is the first choice primarily because, based on the

present state-of-the-art, it is the most reliable. It is therefore recommended for the initial OLO vehicles. The second choice is the storable bi-propellant system. This system has a weight advantage over a stored gas system, but is not yet as reliable. It is estimated that within two years improvements in the state-of-the-art will make this the optimum system. The storable bi-propellant system is described in paragraph b. below.

Cold gas attitude control systems are in use in a number of space applications. Providing good design practice is used, they are inherently reliable, since the large background of hydraulic and pneumatic valve design and test can be applied directly to these systems without extending the state-of-the-art.

Bi-propellant systems, especially for long term storage operation, present new problems in material selection since the propellants are compatible to only a relatively few materials. This is especially true of seal materials. However, a large number of companies have been using these propellants over the last two years not only for attitude control but also as propellants for main boost engines, e.g., first state of Titan II, ICBM. This means that the backlog of information on the use of these propellants is increasing at a rapid rate and will continue over the years, especially when the Apollo program gets under way.

The cold gas nitrogen system operates in the following manner: high pressure nitrogen is stored at 5000 psi in a fibre glass container. The system is energized by a squib operated stop valve, and the nitrogen pressure is reduced to 400 psi by a pressure reducing valve. A schematic diagram of the OLF cold nitrogen system is shown in figure 4-4.

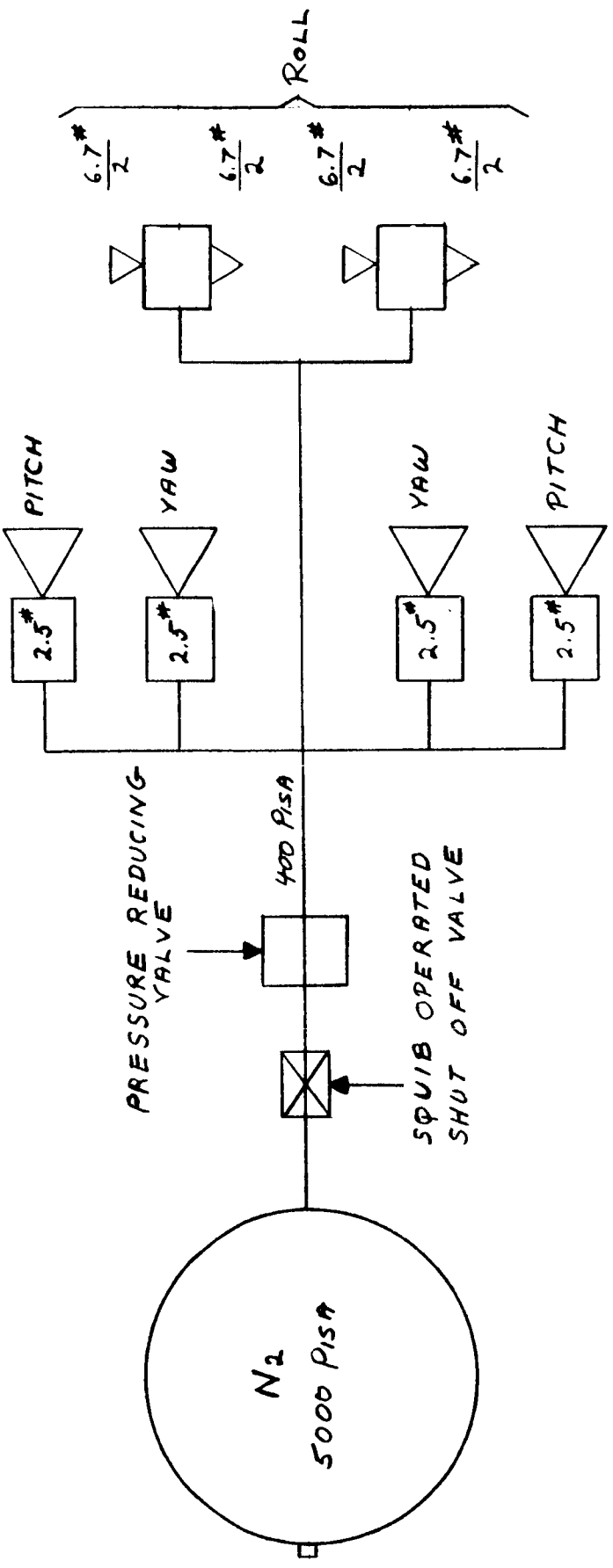
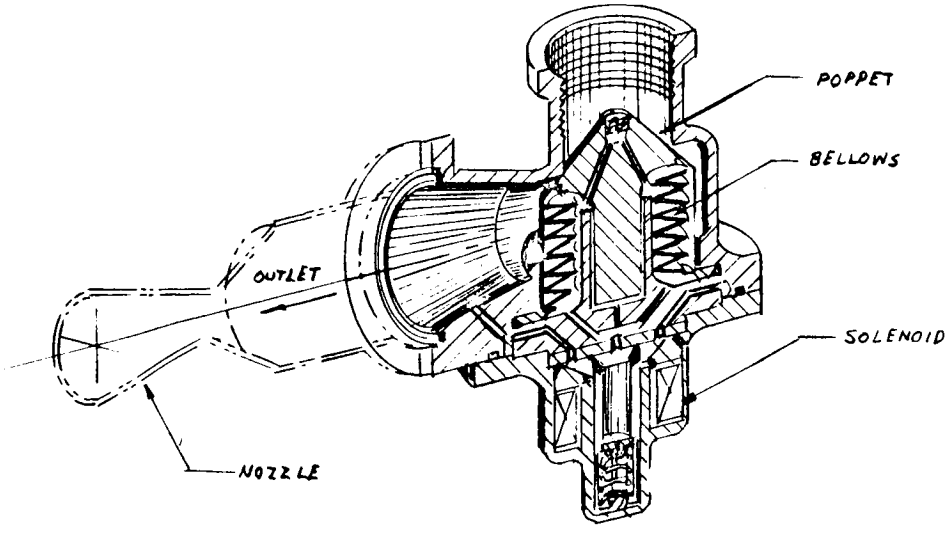


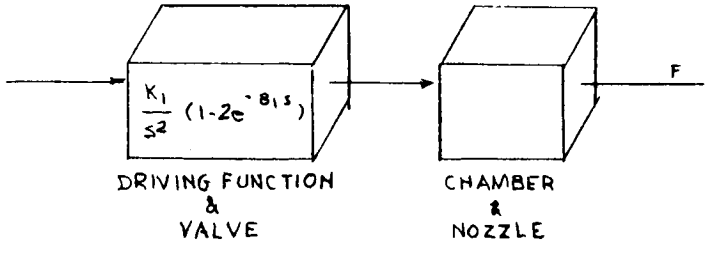

FIGURE 4-4 OLF COLD-NITROGEN SYSTEM

The control valve and nozzle package is shown in figure 4-5. The valve is energized by a pulse width modulated signal. This approach employs a solenoid-staged poppet valve that has no external leakage and develops high closing forces due to the area unbalance. The response time of this system can be adjusted by matching the valve orifice area with the solenoid orifice area. The best design configuration will minimize the gas volume behind the bellows. The position vs time curve for this valve will resemble a ramp output with a transfer function as shown in figure 4-5. Inspection of this transfer function reveals that as the pulse width input signal is decreased, the impulse applied to the vehicle will approach the impulse response of the nozzle and chamber.

b. Alternate Thrust Generator: Storable Bi-propellant System

Improvements in the state-of-the-art in the next two years are expected to increase the reliability of the storable bi-propellant system and make it the optimum choice for later OLO vehicles. A schematic diagram of the system is shown in figure 4-6. The fuel and oxidizer are stored in separate lightweight containers. A squib-operated stop valve allows nitrogen to pressurize the propellant storage containers. The nitrogen is stored at 5000 psi and reduced to 400 psi by a reducing valve. The propellant combination is hypergolic and therefore the combustion is spontaneous in the valve combustion chamber and no external ignition device is required. The valve is energized by a pulse width modulated signal. The fuel and oxidizer controls are tied together mechanically to ensure correct oxidizer lead under all conditions. The valve block diagram and a drawing of a typical unit now in operation at Vickers is shown in figure 4-7.




$K_1$ = SLOPE OF THE VALVE RAMP FUNCTION

$a_1$ = INPUT SIGNAL DURATION IN SECONDS

$\tau_n$ = NOZZLE & CHAMBER CHARACTERISTIC TIME

$K_2$ = THRUST TO VALVE POSITION GAIN

F = THRUST

FIGURE 4-5 SOLENOID-STAGED SYSTEM

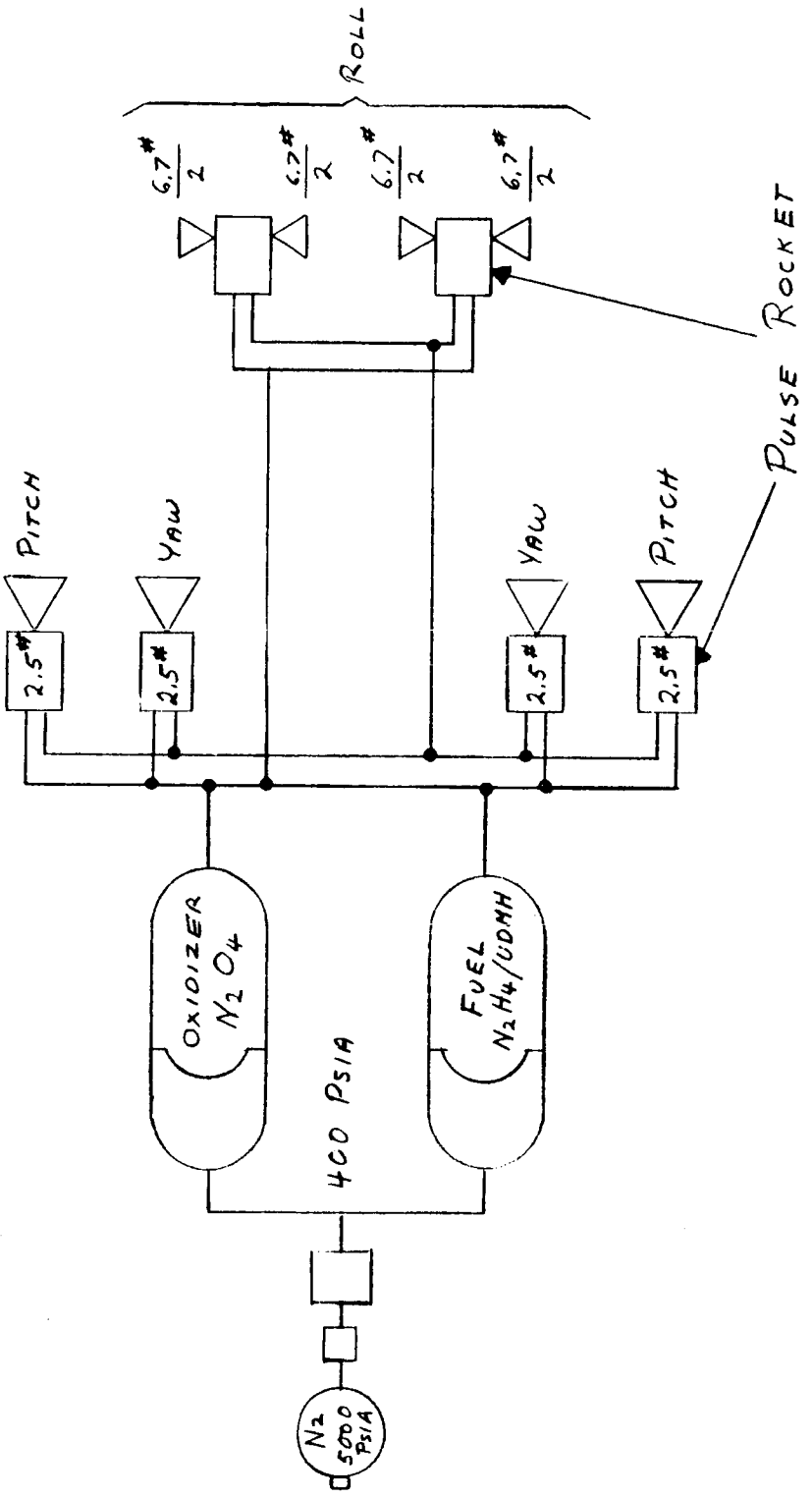
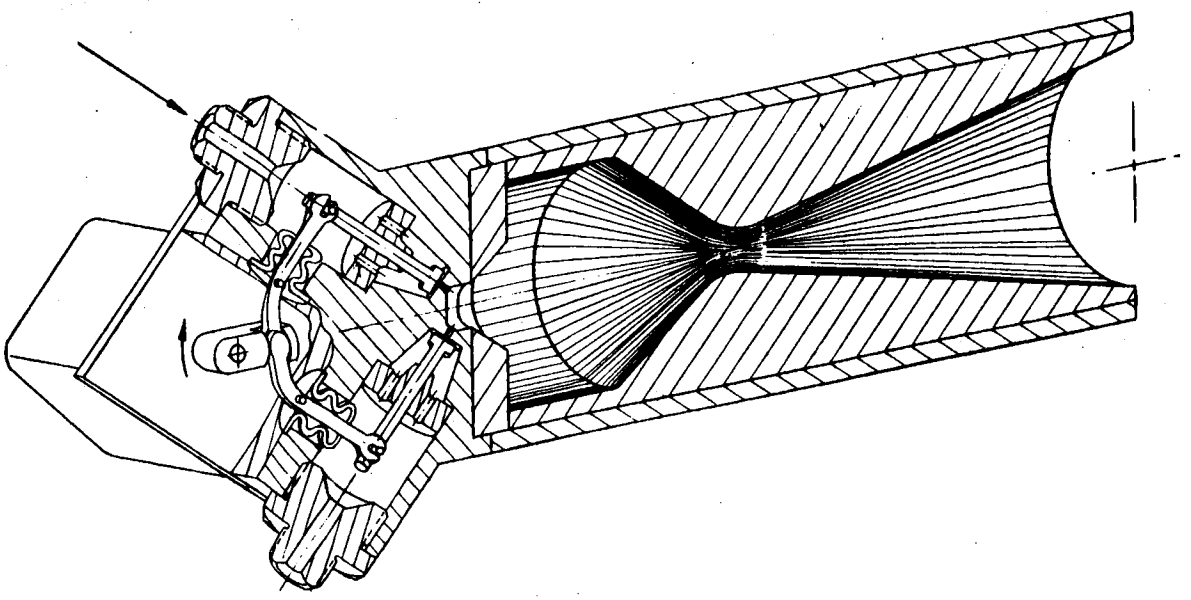
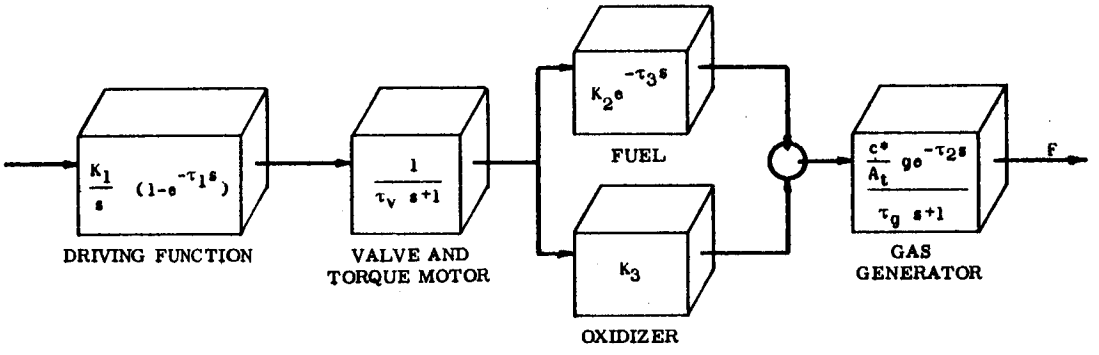



FIGURE 4-6 STORABLE BI-PROPELLANT SYSTEM SCHEMATIC



A-




.008 SEC. – $\tau_v$ = VALVE AND TORQUE MOTOR CHARACTERISTIC TIME

$\tau_1$ = PULSE WIDTH - SECONDS

.002 SEC. $\tau_2$ = TRANSPORT AND COMBUSTION DELAY

.002 SEC. $\tau_3$ = OXIDIZER LEAD AND OVER-RIDE DESIGNED INTO VALVE

.001 SEC. $\tau_g$ = GENERATOR CHARACTERISTIC TIME

$K_1$ = INPUT SIGNAL STRENGTH - AMPS

$\frac{K_2}{K_3}$ = FUEL TO OXIDIZER RATIO = $\frac{1}{1.62}$

$c^*$ = GENERATOR CHARACTERISTIC VELOCITY

$A_t$ = THROAT AREA

$g$ = GRAVITATIONAL CONSTANT

B- BLOCK DIAGRAM WITH PULSE-WIDTH MODULATED SIGNALS

FIGURE 4-7 STORABLE BI-PROPELLANT REACTION-CONTROL SYSTEM

The propellants chosen for the bi-propellant attitude control system are $N_2 O_4$ - Nitrogen Tetroxide, as the oxidizer, and Aerozine 50 as the fuel. The latter is a 50-50 combination of UDMH (unsymmetrical Dimethyl Hydrozine) and Hydrozine, $N_2 H_4$.

Since the vapor of Aerozine 50 is 90 percent UDMH, the toxicity of Aerozine 50 is the same as UDMH. UDMH is fairly toxic, a maximum allowable concentration of 0.5 ppm being recommended. It is not corrosive. A respirator with an ammonia cannister, splashproof goggles and vinyl-coated gloves should be worn. Detection levels by odor are much lower than the level of hazardous concentrations.

Nitrogen tetroxide is classified as a poisonous gas and artificial breathing apparatus is required to handle it. In an OLO application, the oxidizer and control equipment would not be stored in a crew compartment, personnel handling the material outside the vehicle will require breathing apparatus for normal condition.

The fire hazards of Aerozine 50 may be either of the fuel type or the fuel and oxidizer type. The fuel type fire presents the same problems as a gasoline fire, except that firefighting is easier in the case of Aerozine 50 because it is soluble in water. The fuel and oxidizer type fire is much more vigorous but it is controllable by the use of large volumes of water, and is free of the detonation hazards of the non-hypergolic, immiscible types of propellants.

Nitrogen tetroxide is a stable, non-flammable, shock insensitive material. In the absence of combustible materials, and when it is free of dissolved organic contaminants, there are no fire or explosion hazards associated with its use.

The combination of Aerozine 50 and Nitrogen Tetroxide has been in use at Vickers, Inc. for over one year and no accidents have occurred. A significant point to be noted is that this combination of propellant has been selected for the Apollo capsule attitude control system.

## 4-2. UNMANNED PAYLOAD WITHOUT TUG

### a. Recommended System

#### (1) System Configuration

The unmanned payload will rendezvous with the OLF and dock by means of commands generated in the OLF. During rendezvous, the attitude control system must position the payload in attitude for firing the main propulsion rocket. For docking, the attitude control system must be capable of translating the vehicle laterally without rotation and be capable of positioning the vehicle accurately with respect to the OLF. The control system must, therefore, accept attitude, lateral velocity, and thrust commands, and produce torques and lateral accelerations. In addition, inertial attitude angles and velocities must be measured in the vehicle and transmitted to the OLF so that the proper attitude and lateral velocity commands may be computed.

One axis of the recommended attitude control system configuration for the unmanned payload is shown in figure 4-8. Both attitude and lateral vernier velocity control is provided, but the velocity control is used only for docking. The attitude control loop will produce an angular rate proportional to the input signal telemetered from the OLF, and will hold the existing attitude in the absence of an input. The lateral velocity control loop will produce a lateral velocity

proportional to the input from the OLF and will hold essentially zero velocity in the absence of an input. An inertial reference unit is used to measure the inertial attitude and velocity of the vehicle for transmission to the OLF. A transponder, which is used to aid radar tracking of the payload from the OLF, is also shown in figure 4-8.

The equipment required in the OLF to derive commands for rendezvous and docking of the payload is shown in figure 4-9. Although the mode of operation shown is automatic with manual override, it may also be completely manual or manual with automatic assists. The selection of a specific control method will be based on how well the human operator can perform the rendezvous and docking maneuvers in terms of time, extra fuel, and the amount of automatic equipment replaced by him in the OLF.

If manual control is used, the operator in the OLF will have a display of the payload's position, orientation, and angular rates with respect to both the OLF and inertial space. From these quantities he will determine the proper attitude rate and thrust on-off commands to accomplish a rendezvous. He will command a sequence of maneuvers in which the vehicle is alternately oriented to direct the main thrust so as to stop the rotation of the OLF/payload line of sight in inertial space, and then reoriented to direct the thrust so as to follow a certain range-range rate program. For docking, accurate payload position and orientation with respect to the OLF docking centerline will be displayed and the operator will determine the proper payload angular rates and lateral velocity commands.

If automatic control is used, the human operator will be replaced by a computer operating on essentially the

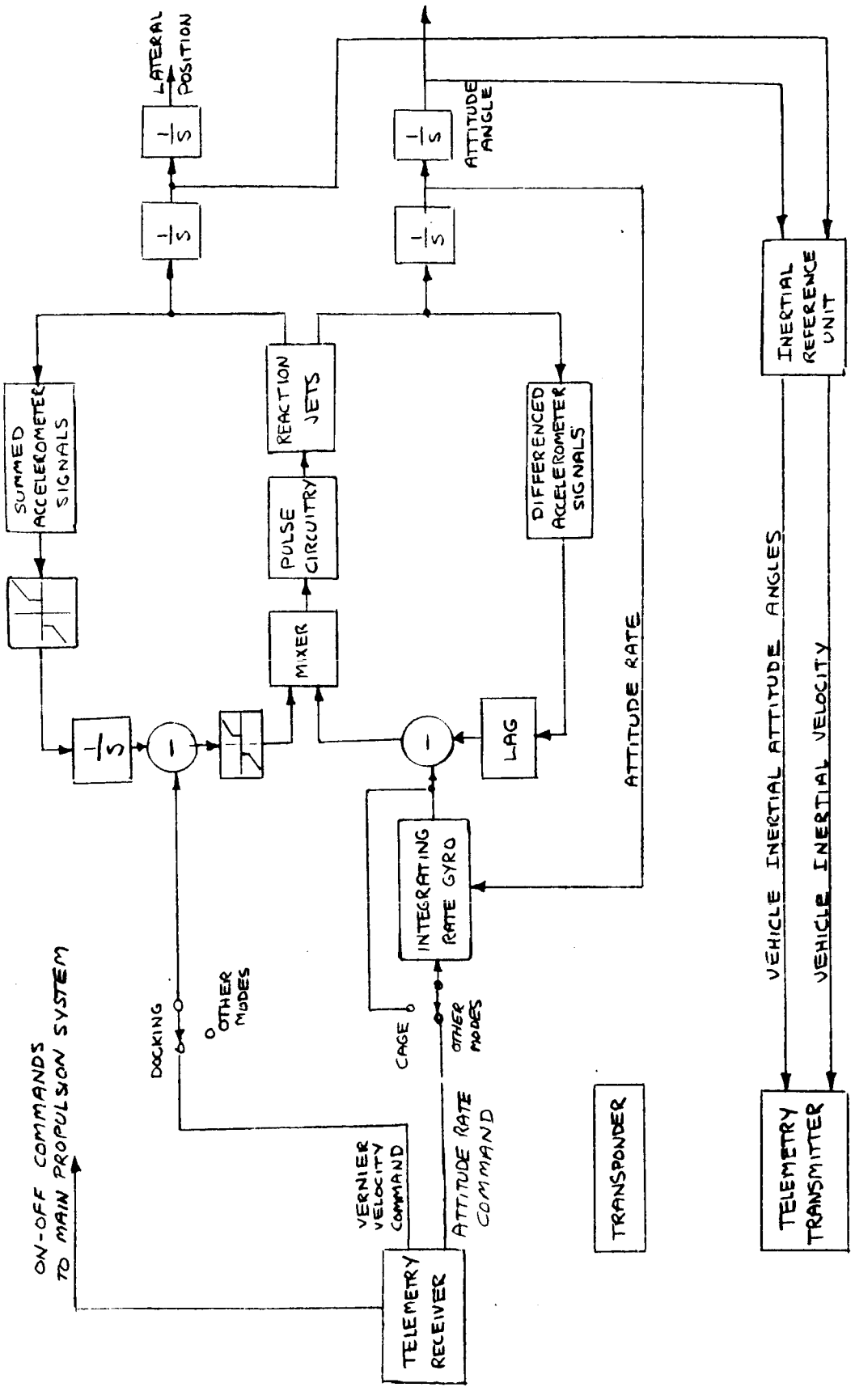



FIGURE 4-8 UNMANNED PAYLOAD ATTITUDE AND VERNIER-VELOCITY CONTROL SYSTEM (NO TUG)

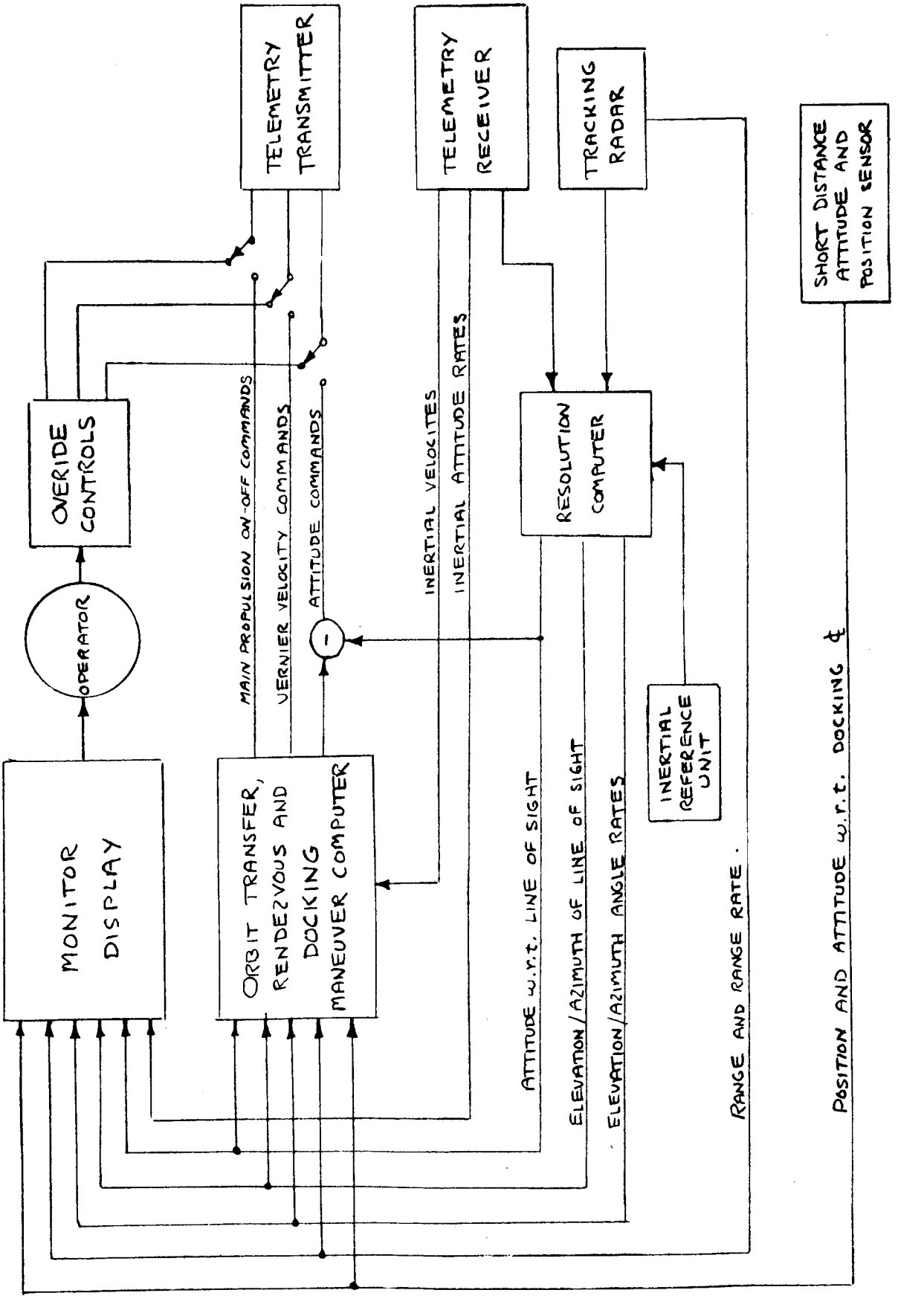



FIGURE 4-9 OLF EQUIPMENT FOR AUTOMATIC CONTROL OF UNMANNED PAYLOADS

same input information. A significant difference will be that the computer will allow more efficient (in terms of time and fuel) rendezvous and docking maneuvers.

(2) System Operation

The torque generators for both the attitude and lateral velocity control loops will be pairs of fixed-thrust reaction jets of the type described in paragraph 4-1. Torques about the vehicle axes are produced by two pairs each of pitch, roll, and yaw reaction jets. Accelerations along the vehicle axes will be produced by a pair of the pitch or yaw reaction jets firing in the same direction and by an additional pair of fore-aft reaction jets. The signal mixer and pulse circuitry shown in figure 4-8 determine the selection of reaction jets and the impulse duration to produce the desired angular on lateral acceleration.

The body mounted integrating rate gyros provide a "floating" attitude displacement reference. The operation of the IRG is illustrated in figure 4-10. The IRG is sensitive to vehicle rates about its input axes and produces a displacement of its output axis proportional to the angular rate. The gyro can also be torqued about its output axis. This results in an output rate proportional to the input (torquing) signal, which represents an angular rate command. The gyro output is proportional to the integral of the difference between the commanded and actual angular rate and is therefore a displacement error.

An angular rate damping signal is derived by passing angular acceleration signals, obtained from accelerometers, through a lag network. The manner in which the outputs of two linear accelerometers can be combined to measure angular

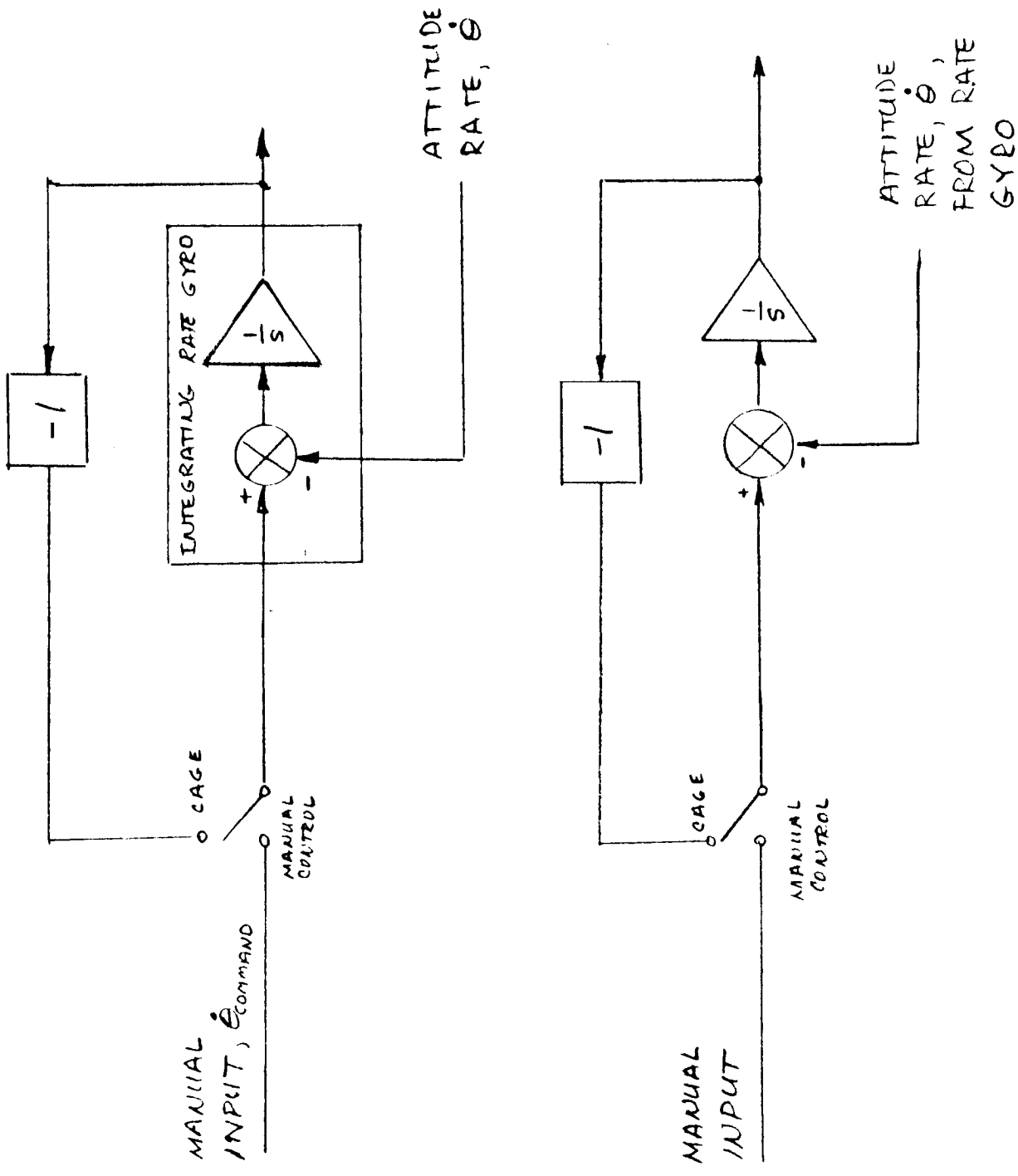



FIGURE 4-10 DRIVEN DISPLACEMENT CONTROL SYSTEM WITH INTEGRATING RATE GYRO VS ELECTROMECHANICAL INTEGRATOR

acceleration is shown in figure 4-11. Linear accelerometers have been suggested as angular acceleration sensors because they are already in the vehicle being required for lateral path-damping signals, and because they are more reliable than rate gyros.

Initial orientation of the payload before it is acquired by the OLF is accomplished by having the IRG follow-up on itself (electrical caging mode) until just prior to booster separation. The payload will then hold the attitude existing at separation.

In the lateral velocity control loop, velocity is measured by integrating the sum of the two linear accelerometer signals mentioned above. Figure 4-11 also shows how the accelerometer signals can be combined to measure the linear velocity of a point in the vehicle. The error between the commanded and measured velocity results in a proportional lateral acceleration.

It should be noted that during rendezvous, the velocity control loop is open and the payload will rotate about its center of gravity in response to attitude commands. When docking, however, both loops are operating and the summed accelerometer signal in the velocity loop will sense the velocity of a point whose location is determined by the manner in which the signals are summed (see figure 4-11). The velocity loop will prevent this point from translating by redistributing the torque commands called for by the attitude loop. This will cause the vehicle to rotate about the selected point rather than the center of gravity. The ability to force the payload to rotate about an arbitrary point which is obtained in this manner may be useful in the docking maneuver.

To accomplish a rendezvous by manual control from the OLF, the operator will generate attitude rate commands

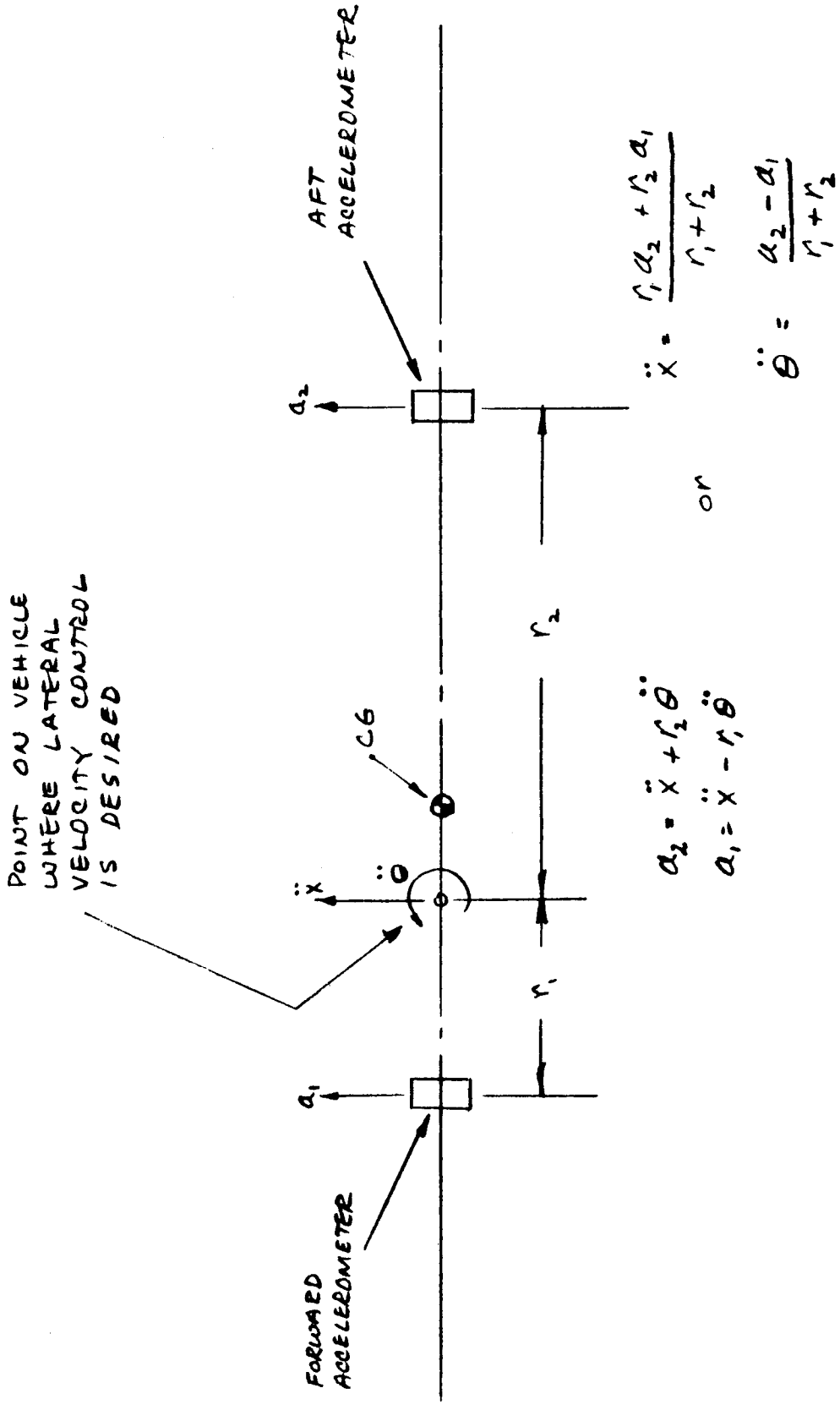



FIGURE 4-11. SCHEMATIC ARRANGEMENT OF LATERAL ACCELEROMETERS FOR MEASURING ANGULAR AND LINEAR ACCELERATIONS

and rotate the payload until it is at the proper attitude for firing the main rocket as indicated by his display of payload attitude with respect to the line of sight, measured in payload body axes. These angles are obtained by combining the payload and OLF inertial orientation angles with the tracking radar orientation with respect to the OLF. The proper attitude for firing, and the required thrust duration, are determined by the other displayed quantities and the rendezvous technique being used. Some of the displayed quantities shown in figure 4-9 will probably be pre-digested automatically to simplify the operator's task. The payload may be docked using a display of payload position and orientation with respect to the OLF docking centerline, or by direct visual observation. The operator will generate attitude rate and lateral velocity commands.

With automatic control of rendezvous from the OLF, the proper payload attitude for thrusting will be determined by a computer. Payload attitude will be controlled in a closed loop manner by comparing the payload orientation with respect to the line of sight to the computed orientation. The attitude error will be transmitted to the payload as an attitude rate command.

For automatic docking, closed loop control of payload position with respect to the OLF docking centerline will be provided. Position errors measured in payload body axis will be transmitted to the payload as lateral velocity commands.

(3) Sensors

(a) Integrating Rate Gyro

The performance specifications of a typical IRG are listed in paragraph 4-1. This gyro will satisfy the requirements

of the unmanned payload control system shown in figure 3-2. Since the drift rate is less than 0.01 deg/hr., the IRG is also a satisfactory long period displacement reference for the attitude control loop prior to OLF acquisition.

(b) Linear Accelerometers

Preliminary calculations indicate that the linear accelerometers used to measure angular and linear accelerations can be of the E-pickoff type. It is true that a null output error will result in an attitude angle error and a residual lateral acceleration, and that any mismatch of accelerometer gradients will result in cross-coupling effects. For example, lateral velocity error will arise from a constant angular rate and an attitude angle error will be caused by a constant lateral acceleration. However, the magnitudes of these effects have been calculated and appear to be negligible.

The accelerometer null output will require an IRG output and therefore a positioning error to cancel it. A null output of 1% of full scale will cause an attitude hang-off of approximately .1 deg., which is within the positioning accuracy tolerances. The accelerometer null output also enters the velocity control loop and results in a steady state acceleration. Thus the null error must be removed by deadzones on the accelerometer output and on the velocity error, as shown in figure 4-8. This, in turn, will impose a velocity threshold on the system. The minimum value of this threshold is a function of the dynamic response of the accelerometer-integrator combination (not shown), and the characteristics of the pulse modulation circuitry and reaction jets. The velocity threshold of 0.01 fps called for in the table of figure 3-2 for the docking mode appears to be

unrealistic and some difficulty is anticipated in achieving it. If the threshold specification cannot be met using accelerometers, an alternate solution to the lateral control problem is required. One possibility is to provide closed loop control of payload lateral position. The short range position sensor would serve as a position reference and derived lateral velocity would be used for damping.

The cross-coupling errors which are caused by mismatched accelerometer gradients have also been estimated. With a constant angular rate of 12 degrees per second, a 1 percent mismatch in gradients will cause a 0.02 fps velocity standoff. With a constant linear acceleration of 10 $fps^2$, a 1 percent mismatch will cause a 3-degree attitude error. Cross-coupling errors are not expected to be a problem in the docking maneuver, since the angular rates and lateral accelerations are low and are of short duration.

(4) Torque Generators

Four types of reaction-jet systems were considered for the unmanned payload. An analysis of the unmanned payload requirements resulted in the following weight breakdown.

| Type of System | Propellant Weight | Tare Weight | System Weight |
|---|---|---|---|
| Cold Gas-Nitrogen | 139.35 lbs | 263.65 lbs | 403.0 lbs |
| Storable Bi-propellant | 32.8 lbs | 119.2 lbs | 152.0 lbs |

The systems were evaluated using the same selection parameters and weighting factors as for the OLF. (See paragraph 4-1, a.) The results of the trade-offs are as follows:

First Choice - Cold Nitrogen
Second Choice - Storable Bi-propellant

Here again the cold Nitrogen system is the first choice, primarily because of its present reliability advantage. Within two years, however, improvements in the state-of-the-art are expected to make the storable bi-propellant system the optimum choice.

b. Alternate Systems and Trade-offs

(1) Alternate Sensors

Alternate sensors that have been considered for use in the payload attitude control loop are rate gyros in place of the accelerometers and a stable platform attitude reference in place of the IRG.

A rate gyro can be used in place of the accelerometers and lag network to provide an attitude rate damping signal. The two techniques are dynamically equivalent. The factors involved in the choice of sensor are as follows:

- Accelerometers are required for lateral path damping and are already in the system.
- Calculations show that simple E-pickoff type accelerometers will probably be satisfactory. These are more reliable than rate gyros.
- The problems that arise in measuring aircraft angular acceleration with linear accelerometers, due primarily to gradient mismatch and null imbalance, do not appear to be significant in space. The difference is due to the absence of vibration and the low range required.

- The use of rate gyros will eliminate the cross-coupling errors between the lateral velocity and attitude control loops. At the moment these do not appear to be a problem.

The inertial attitude angle reference (platform) already in the payload can conceivably be used in place of the IRG. However, resolvers will be required to derive body axis angles from the inertial angles, and a synchronizing integrator will also be required in each axis to synchronize the control system with the resolved platform outputs prior to initiation of remote control. Manual control of the payload from the OLF would be the same, with the operator generating angular rate commands. These would be fed through the synchronizing integrators to give a "driven displacement" system similar to that obtained with the IRG's. The factors involved in the choice of IRG sensors are as follows:

- The use of the IRG eliminates the need for resolvers and synchronizers in the payload and places them in the OLF where they can be used for each payload.
- The IRG provides initial orientation of the payload when placed in follow-up prior to booster separation.

## 4-3. SPACE VEHICLE (S/V) NO TUG

In the absence of tug assistance, the S/V must be taxied to a launch site some distance from the OLF and oriented for launch. This involves orientation of the vehicle to thrust away from the OLF, coast to the launch site, reorientation for braking at the launch site, and orientation for launch. The latter maneuver may be controlled from the OLF in the same manner as the unmanned payload in order to take advantage of its centralized sensing and computing facilities. The control system requirements will be similar to those of the unmanned payload and will not be discussed in detail here. (See paragraph 4-2.)

**The torque generators for the Space Vehicle** will operate on hydrogen from the main propulsion tanks. This is the same system as is recommended for the astrotug. The system is described in paragraph 4-4, a, (4), and shown in figure 4-12.

## 4-4. MANNED ASTROTUG

### a. Recommended System

#### (1) System Configuration

The astrotug's functions include rendezvous and docking with the incoming payload, followed by rendezvous and docking at the OLF with the payload in tow. The rendezvous technique assumed is a simple one based on the results of NASA simulator studies as described in paragraph 3-1,d. Control of tug attitude will be required for rendezvous, while control of both attitude and lateral velocity will be

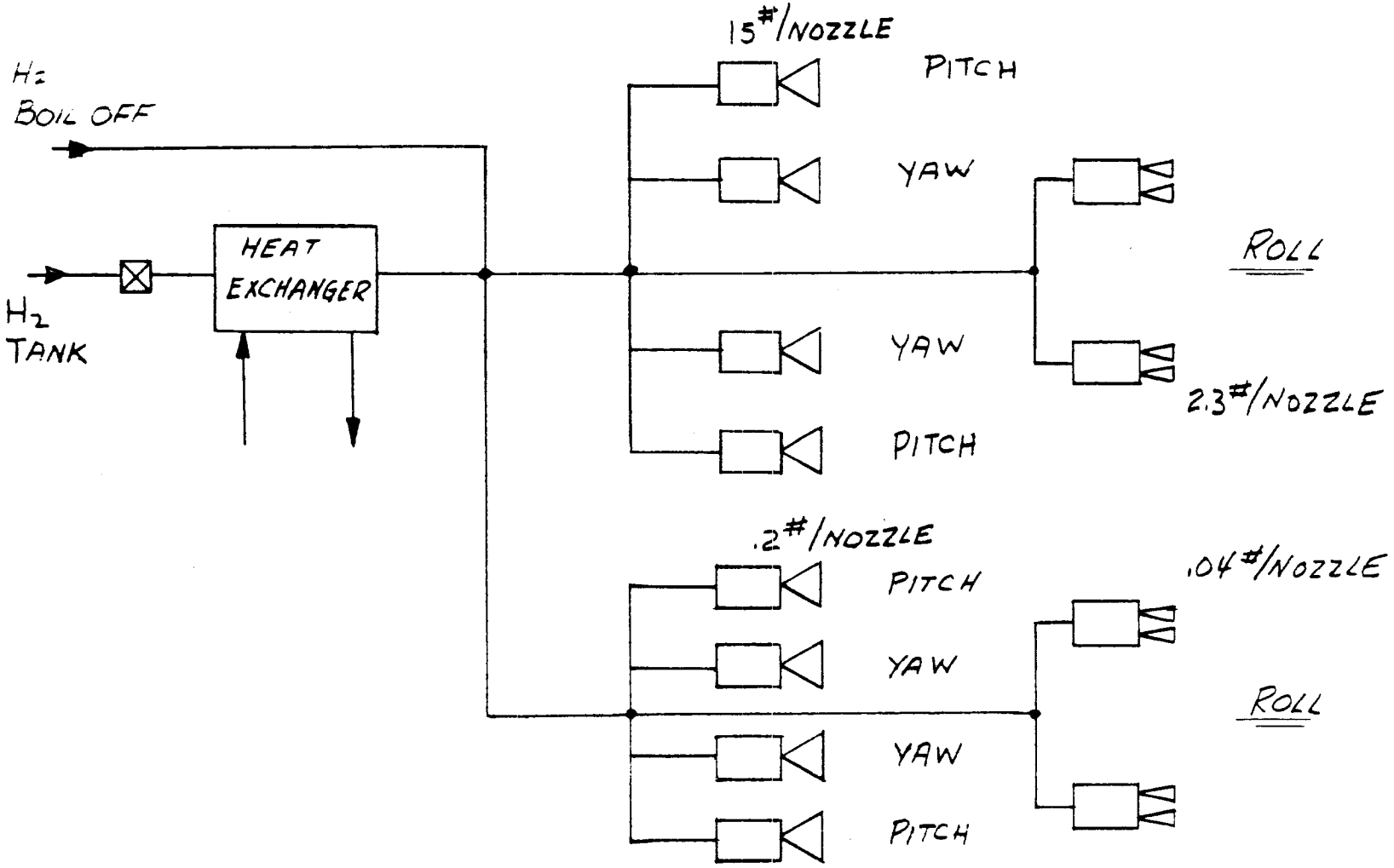



FIGURE 4-12 S/V SCHEMATIC

WEIGHT ESTIMATE (SINGLE SYSTEM : NO REDUNDANCY) = 42 LBS.

required for docking. During payload return, the payload attitude control system may or may not be coupled to the tug control system. The system configurations discussed below cover both possibilities.

One axis of the recommended system configuration for the manned astrotug is shown in figure 4-13. The system provides control of attitude rate and lateral acceleration. The attitude control loop will produce a vehicle rate proportional to the pilots input command and will provide short period attitude hold in the absence of an input. This basic, or minimum, system places the maximum workload on the pilot. The pilot must also be provided with a display (not shown) of tug position, orientation, and angular rates with respect to the payload or OLF and to inertial space. An inertial reference unit and a tracking radar will be required to provide information for the display. With these quantities the pilot can determine the proper attitude rate and thrust on-off commands to accomplish a rendezvous. Optical references will be used in docking on the payload and OLF.

The ability of the pilot to accomplish efficient rendezvous and docking maneuvers with the minimum system is questionable. An augmented tug control system is shown in figure 4-14 which will provide pilot relief as well as more precise control by using additional equipment. Both attitude angle and lateral velocity control loops are provided. The attitude control loop will accept attitude rate commands from the pilot and hold the existing attitude angle accurately in the absence of a command. The lateral velocity control loop will accept lateral velocity commands and hold essentially zero velocity in the absence of a command.

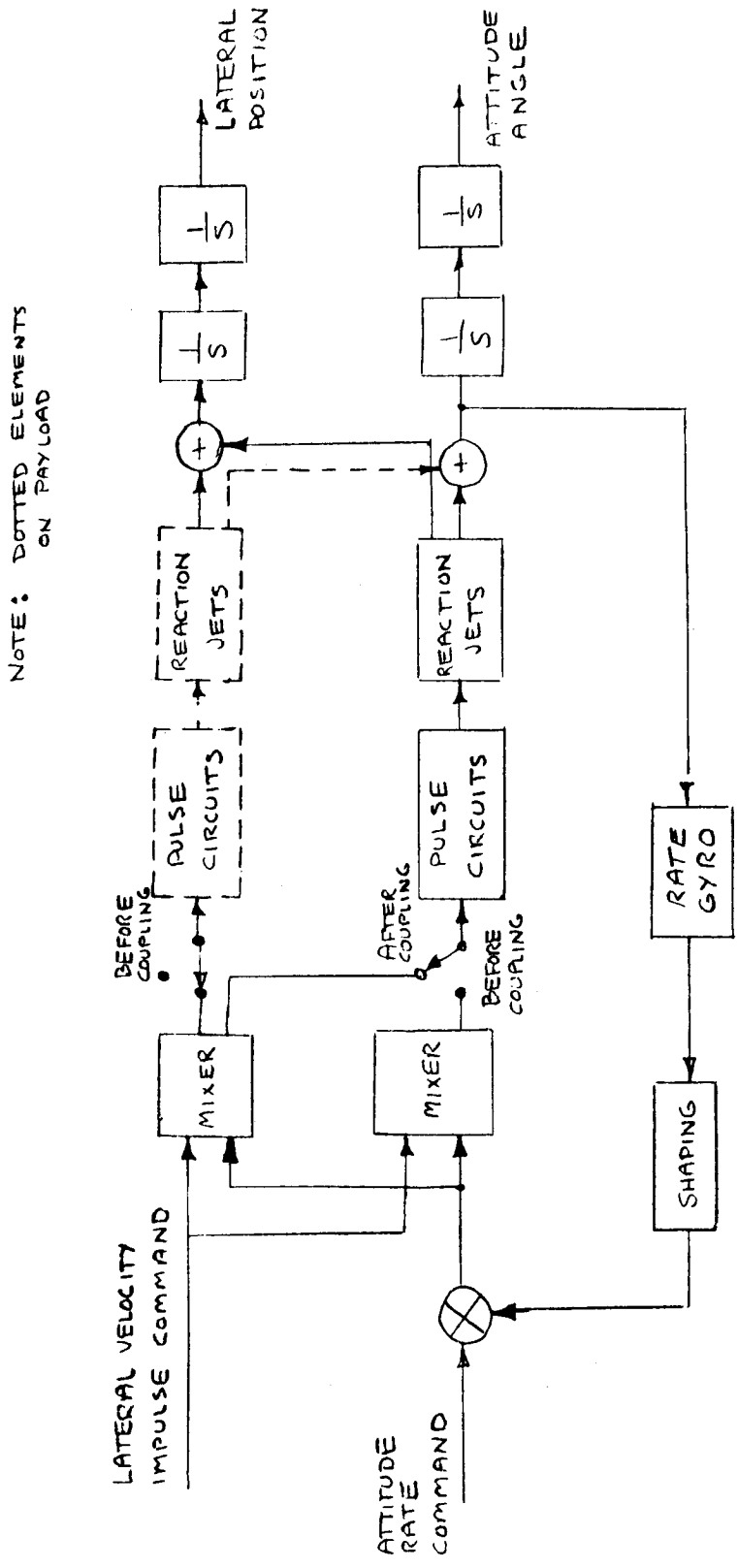



FIGURE 4-13 MINIMUM ATTITUDE-CONTROL SYSTEM FOR MANNED ASTROTUG

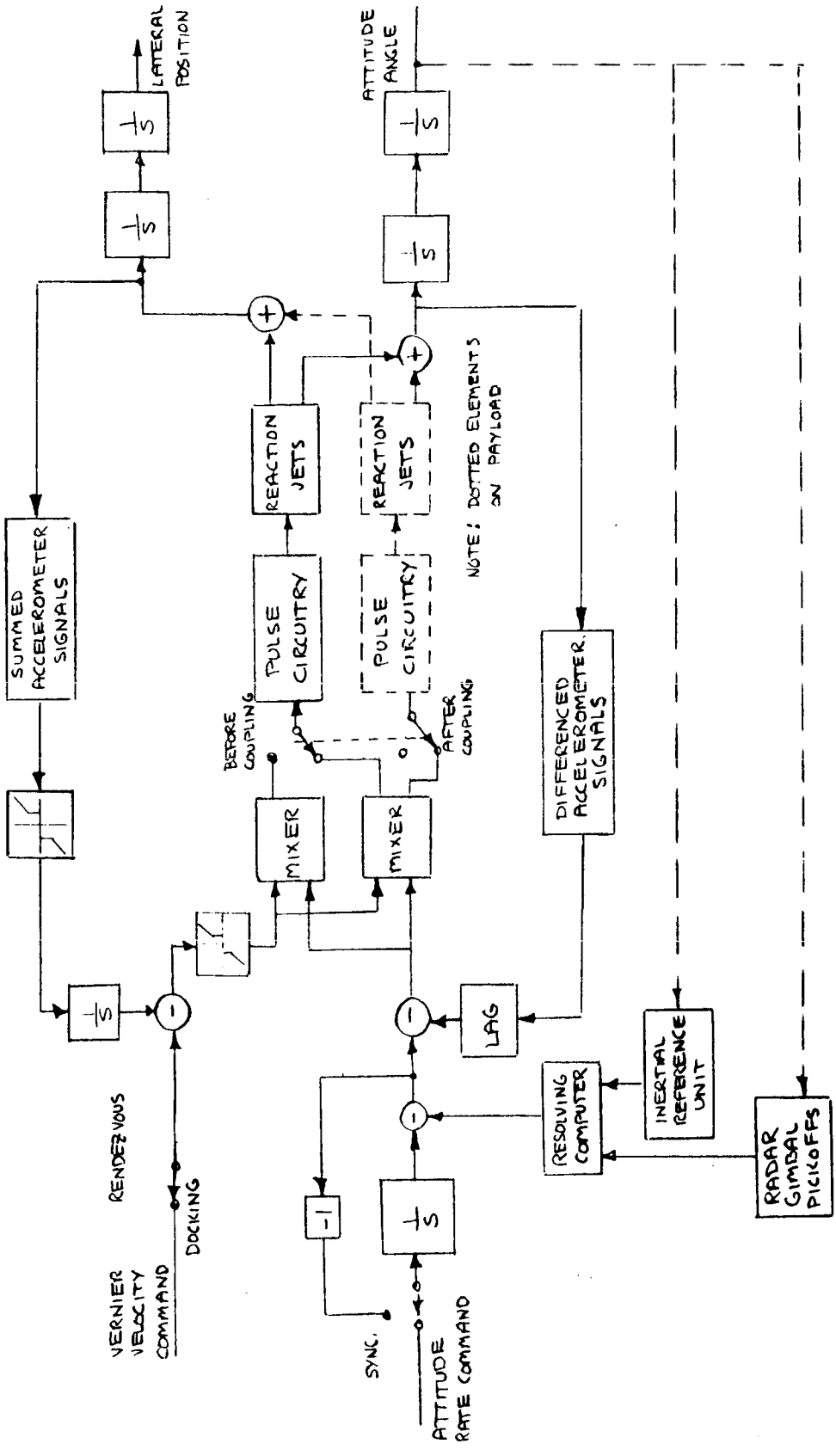



FIGURE 4-14 AUGMENTED ATTITUDE AND VERNIER-VELOCITY CONTROL SYSTEM FOR MANNED ASTROTUG

(2) System Operation

The torque generators for the tug control system will be fixed thrust reaction jets of the type described in paragraph 4-1. The signal mixers and pulse circuitry shown in figures 4-13 and 4-14 determine the selection of reaction jets and the impulse duration to produce the desired angular and lateral acceleration.

In the minimum system rate gyros are used to sense angular rates in body axes. The difference between the measured angular rate and the input signal commands a proportional torque. To obtain a lateral velocity, the pilot applies a lateral velocity impulse, that is, he determines the duration of the lateral thrust directly and closes the loop through his display.

The augmented tug control system is similar to unmanned payload system described in paragraph 4-2. Both systems make use of summed and differenced linear accelerometer signals to measure lateral and angular accelerations. The two systems differ only in the method of deriving an angular displacement error and of inserting commands into the attitude loop. The tug attitude control system derives the attitude of the vehicle by resolving signals from an inertial reference unit and radar gimbal pickoffs. Synchronizing integrators are required to assure a zero attitude angle error signal prior to engaging the attitude control system. In normal operation, attitude rate commands are inserted through the synchronizing integrators. This results in a "driven displacement" type of control.

The operation of the velocity control loop and the interaction of the velocity and attitude control loops during

docking is identical to that of the unmanned payload system and is described in paragraph 4-2,a,(6).

(3) Sensors

(a) Rate Gyro

The performance specifications of a typical 3 axis rate gyro assembly are listed in paragraph 4-1. In the minimum system configuration, this gyro would satisfy the requirements of the tug control system shown in figure 3-3. With a gyro null of 0.03 deg/sec. it will take 16 seconds to exceed the 0.5 deg. positioning accuracy specification. This appears satisfactory for short period attitude hold. The threshold rate of 0.01 deg/sec. is within the specification.

(b) Linear Accelerometers

Accelerometers are discussed in paragraph 4-2,a,(3).

(c) Inertial Reference Unit

The inertial reference unit is required in the tug to supply information for the pilot's display for navigation during the rendezvous maneuver. Its direct use in the augmented attitude control system is discussed in the next section. No difficulty is anticipated in meeting the attitude control specifications with this sensor.

(4) Torque Generators

A schematic diagram of the reaction jet system recommended for the manned astrotug is similar to that shown for the S/V. The system operates on hydrogen from the main propulsion tanks, assuming that this is one of the propellants

used for the tug thrust engine. For the majority of operations, there will be an adequate supply of boiloff from the cyrogenic hydrogen storage. However, for a condition where additional hydrogen is required, a supply of liquid hydrogen will be taken from the main tank and flashed in a heat exchanger operating on waste heat from the crew or electronic compartments. A gas pressure of 10 psi and a gas temperature of 460°R are assumed for this system.

There are two sets of nozzles for control about each axis. One set will produce high torque levels for payload rendezvous and return, the other set will produce low torques for the payload coupling. The maximum thrusts required per nozzle for the four tug functions are tabulated below.

| | Payload Rendezvous | Payload Coupling | Payload Return (No Help | Payload Docking From Payload) |
|---|---|---|---|---|
| Max. Thrust per nozzle | Pitch 73.4 | 1.0 | 133.0 | 26.7 |
| | Yaw 73.4 | 1.0 | 133.0 | 26.7 |
| | Roll 40.0 | 0.3 | 80.0 | 8.0 |

Due to the variation in thrust levels required for the four tug functions, a variable throat area nozzle is proposed to provide the three highest values shown above. A small unit will be used for payload coupling.

The weight breakdown for the system is as follows:

| | |
|---|---|
| System tare weight | 113 lbs |
| Propellant weight chargeable to attitude control | 77 lbs |
| Total System Weight | 190 lbs |

b. Alternate Systems and Tradeoffs

(1) Alternate Configurations

The two configurations considered for the astrotug, that is, the minimum and augmented systems, have already been described. The minimum system may be all that is needed for rendezvous and docking and has the advantage of being simpler and probably more reliable. The advantages of the augmented system are as follows:

- Control of attitude displacement through a long period displacement reference is more accurate and provides more pilot relief than short period control through a rate reference, particularly in the docking maneuver.
- Closed loop control of lateral velocity is more accurate and provides more pilot relief than open loop control.
- The "driven displacement command" type of control combines a desirable maneuver characteristic (rate inputs from the pilot) with accurate attitude displacement control.

(2) Alternate Sensors

Rate gyros can be used in place of accelerometers in the attitude loop. This alternate was discussed in paragraph 4-2,b,(1).

Integrating rate gyros can be used in place of the inertial reference, radar pickoffs, resolving computer, and synchronizers to provide a displacement reference for the attitude loop. This tradeoff was evaluated for the unmanned

payload control system with the result being the choice of the IRG. The factors involved in the choice of the inertial reference unit, et. al., for the astrotug are as follows:

- The inertial reference unit radar gimbal pickoffs, and resolving computer are already required on the astrotug for driving the pilot's display. (This was not the case in the unmanned payload tradeoff).
- With "driven displacement" control, the output of the synchronizing integrator, which is the actual displacement command, can be displayed permitting a more precise control of attitude. The displacement command can be positioned by the pilot and the system will follow-up on it with no error.

## 4-5. UNMANNED PAYLOAD (WITH TUG)

### a. Recommended System

#### (1) System Configuration

When tug assistance is available, the unmanned payload control system is required to damp the initial tumbling and upon command from the tug, hold its attitude while the tug attaches itself. The control system may also be required to accept thrust commands from the tug in the event that the payload has to provide all or part of the attitude and/or propulsion forces required for the return trip and docking.

The recommended control system configuration for the payload is shown in figure 4-15. The torque generators and the associated control circuitry is similar to that described for the other vehicles. Integrating rate gyros are used as the attitude reference.

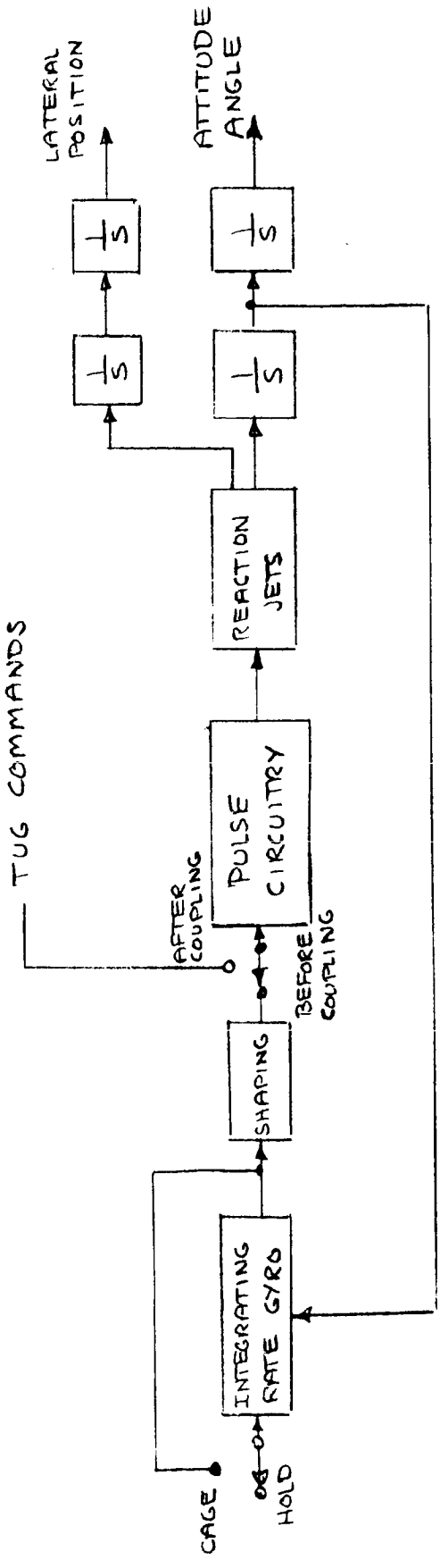



FIGURE 4-15 UNMANNED PAYLOAD ATTITUDE-CONTROL SYSTEM (WITH ASTROTUG)

(2) System Operation

The integrating rate gyro is sensitive to body axis rates and produces an error signal proportional to the integral of the rate, that is, a displacement error. The IRG is caged during the injection phase and released just prior to booster decoupling. This establishes a reference attitude which the control system will hold while waiting for the tug to arrive. The shaping network is used to provide damping in the loop.

Two modes of operation are provided in the control system. One is an equivalent "low gain" mode for the period before tug arrival when it is only necessary to hold attitude to ± 5 degrees (see figure 3-4). The second is a high gain mode for the tug coupling maneuver when it is necessary to hold attitude to ± 0.5 degree. The two modes may be implemented by changing the error threshold in the pulse circuitry on command from the tug. The use of an equivalent low gain mode will reduce the weight of propellant required for attitude hold.

(3) Sensors

(a) Integrating Rate Gyro

The performance specifications of a typical IRG are listed in paragraph 4-1. This gyro satisfies the requirements of the unmanned payload control system shown in figure 3-4. With a drift rate of 0.01 degree per hour it will require 500 hours to exceed the 5-degree positioning-accuracy requirement of the initial alignment mode.

b. Alternate Systems and Trade-Offs

(1) Alternate Configurations

During payload return it may be advantageous to couple the payload reaction jets into the tug control system to provide all or part of the attitude control and/or propulsion forces required. The factors involved in evaluating these alternatives are the torque and thrust requirements and the fuel requirements for the payload and tug. The tug and payload requirements are shown in figures 3-3 and 3-4 respectively. The pertinent fuel trade-off comparisons are as follows:

- The maximum torque output required from the tug control system is almost doubled (4000 to 2200 ft-lbs) if there is no coupling. (See figure 3-3).
- If coupling exists the payload system must provide all or part of 4000 ft-lbs maximum compared to 26 ft-lbs with no coupling (figure 3-4).
- The maximum tug thrust required (4000 lbs) is the same with or without thrust augmentation from the payload.
- If payload thrust augmentation is used the payload must provide all or part of 4000 lbs thrust, compared to no requirements with no thrust augmentation.
- With no payload assistance the tug uses 400 lbs more fuel than with assistance.
- With payload assistance the payload uses 300 lbs more fuel than with no assistance.

From the above comparisons it can be seen that the tug payload coupling question involves a number of other factors including the design of the torque and thrust generators of both vehicles and cannot be answered at this time.

(2) Alternate Sensors

An inertial reference unit could be used in place of the three integrating rate gyros in the payload control system. However, this alternate has nothing to recommend it. The IRG's are satisfactory and also enjoy a considerable size and weight advantage.

A rate gyro and an electromechanical integrator could also be used in place of an IRG. This alternate has a size, weight, and reliability disadvantage. The combined effect of the rate gyro and integrator null outputs in cascade gives it an accuracy disadvantage as well.

## 4-6. SPACE VEHICLE (WITH TUG)

### a. Recommended System

The recommended system for the OLV when a tug is available is the same as that used for the unmanned payload. The discussion in paragraph 4-5 also applies to the OLV.

# SECTION V

# COST AND SCHEDULE

## 5-1. GENERAL

A summary of the estimated cost of developing and procuring attitude-control equipment hardware for OLO vehicles through 1966 is given in this section.

The program and expenditures recommended for calendar year 1962 are discussed in detail in Section VI.

## 5-2. BASIS FOR ESTIMATE

Development cost estimates are based on a generalized four year program plan for all vehicles. Major milestones in this program include the following:

First Year - Systems study and design reliability evaluation

Establishment of system specifications.

Second Year - Preparation of engineering models

Performance Evaluation of system models

Reliability evaluation test

Environmental evaluation tests

Component Design

Product Follow-up

Design of Support Equipment

Third Year - Product Follow-up

Preparation of operating and service instructions

Formal Environmental Tests

Formal Reliability Evaluation Tests

Field Test Follow-up

Fourth Year - Product modifications

Product Follow-up

Reliability Evaluation and Follow-up

Field Tests Follow-up

The production schedule consists of delivery of the first engineering prototype early in the third year, with further prototypes delivered throughout the third year. The first production model would be delivered for field testing early in the fourth year.

Hardware procurement cost estimates were obtained by comparison with production hardware of similar complexity. It should be noted that both development and procurement costs reflect the heavy emphasis placed on reliability. The proposed reliability program is discussed in Section VI.

Development and procurement expenditures are scheduled in accordance with the "typical program" outlined by Vought Astronautics. The major milestones in this program are

| | |
|---|---|
| Circumlunar Mission | March 1966 |
| Lunar Orbit Mission | July 1966 |
| Lunar Landing Mission | April 1967 |

Requirements for operational availability of the individual vehicles in the OLO complex have not been established by Vought Astronautics and are beyond the scope of an attitude control system study. The following assumptions were made for purposes of estimating nominal delivery rates:

- Unmanned orbital flight tests of the initial orbiting facility, the orbital launch vehicle, and the initial unmanned payload will be performed in 1964/65 period. These vehicles must attain operational status by the end of 1965 in order that manned tests can be conducted during 1966. At least ten sequentially successful unmanned test flights will be required before a vehicle achieves a man-rated status.
- The first application of the initial orbiting facility, the initial unmanned payload and the initial orbital launch vehicle will be in the lunar orbit missions of mid-1966. The circumlunar mission will be a direct shot with no orbital operations.
- Unmanned orbital flight tests of the intermediate orbiting facility, the intermediate unmanned payload, and the astrotug will be performed during the 1965/66 period. Manned tests will follow in late 1966, with the first application of these vehicles occurring in the lunar landing mission of early 1967.

## 5-3. ESTIMATED COSTS AND SCHEDULE

A summary of attitude control equipment development costs is given in Table 5-1. Several assumptions are implicit in the figure in addition to the basic estimating procedures outlined in the preceding section.

- The cost estimates are based on concurrent development of all systems by Sperry Rand.

- The cost of the orbital launch facility includes development of sensors and attitude controls for use during retrieval and docking. This represents a major OLO problem area.
- The initial orbiting facility and unmanned payload utilize cold nitrogen reaction controls while the intermediate models have storable bi-propellant systems. The quoted price includes the funds necessary to complete development of the storable bi-propellant systems.

A summary of attitude control hardware procurement costs is also shown in table 5-1. The costs are, of course, contingent on the reference OLO schedule given in the table.

Schedules other than the one shown may be established by the operations analysis now underway at Vought Astronautics. If a particular vehicle is not required until a later date than shown, the initiation of development of an attitude control system for this vehicle may be delayed.

Assuming the program length is as shown in table 5-1, the price will increase roughly 4% per year of delay because of price index increases and growth in system complexity. Program stretchouts will raise the total cost, since a basic organization must be supported for a longer time. Development of those vehicle attitude control systems which must be operational in 1965 cannot be delayed. The extensive qualification and flight test program required for manned orbital vehicles makes it imperative that system specifications be established by the end of 1962.

TABLE 5-1

SUMMARY OF ATTITUDE CONTROL EQUIPMENT
COSTS FOR ALL OLO VEHICLES IN THOUSANDS OF DOLLARS

| Calendar Year / Vehicle | 1962 | 1963 | 1964 | 1965 | 1966 | Five Year Total |
|---|---|---|---|---|---|---|
| INDIRECT OPERATING EXPENSES | | | | | | |
| Attitude Control Hardware Development Costs | | | | | | |
| • Orbital Launch Facility | | | | | | |
| Initial | 130 | 500 | 600 | 350 | | 1580 |
| Intermediate | | 110 | 430 | 510 | 300 | 1350 |
| • Space Vehicle | | | | | | |
| Chemical | 90 | 350 | 420 | 250 | | 1110 |
| • Miscellaneous | | | | | | |
| Unmanned Payload | | | | | | |
| Initial | 60 | 220 | 270 | 150 | | 700 |
| Intermediate | | 70 | 270 | 330 | 180 | 850 |
| • Orbital Support Equipment | | | | | | |
| Astrotug | | 130 | 340 | 460 | 290 | 1310 |
| TOTAL INDIRECT OPERATING | 280 | 1380 | 2380 | 2050 | 770 | 690[illegible] |

TABLE 5-1 (Cont)

SUMMARY OF ATTITUDE CONTROL EQUIPMENT
COSTS FOR ALL OLO VEHICLES IN THOUSANDS OF DOLLARS

| Vehicle | Unit Cost, Thousands of Dollars | Rate of Procurement | | | | | Five Year Total Cost |
|---|---|---|---|---|---|---|---|
| DIRECT OPERATING EXPENSES | | 1962 | 1963 | 1964 | 1965 | 1966 | Thousands of Dollars |
| Attitude-Control Hardware Procurement Costs | | | | | | | |
| • Orbital Launch Facility | | | | | | | |
| Initial | 110 | | | AABBBB | CCCCCCCCCC | DE | 2310 |
| Spare Parts | | | | F | F F | | |
| Intermediate | 120 | | | | A A BBBB | C C | 1080 |
| Spare Parts | | | | | F | | |
| • Space Vehicle | | | | | | | |
| Chemical | 75 | | | AABBBB | C C [1] | DE | 900 |
| Spare Parts | | | | F | | F | |

Nomenclature:
A = Ground Test
B = System Ground Test
C = Flight Test
D = Manned Flight Test
E = First Operational Use
F = Spare

Notes:
1 Ten tests not required for man-rating because vehicle is modification of existing vehicle
2 No man-rating required

20-40% spare parts costs during initial evaluation tests is an average price

TABLE 5-1 (Cont)

SUMMARY OF ATTITUDE CONTROL EQUIPMENT
COSTS FOR ALL OLO VEHICLES IN THOUSANDS OF DOLLARS

| Vehicle | Unit Cost, Thousands of Dollars | Rate of Procurement | | | | | Five Year Total Cost |
|---|---|---|---|---|---|---|---|
| DIRECT OPERATING EXPENSES (Cont) | | 1962 | 1963 | 1964 | 1965 | 1966 | Thousands of Dollars |
| • Miscellaneous | | | | | | | |
| Unmanned Payload | 80 | | | | A A BBBB F | C C [2] | 720 |
| Initial Spare Parts | | | | | | | |
| Intermediate Spare Parts | 100 | | | | | AABBBB F | 700 |
| • Orbital Support Equipment | | | | | | | |
| Astrotug | 100 | | | | A A BBBB F | C C [2] | 900 |
| Spare Parts | | | | | | | |
| TOTAL DIRECT OPERATING | | | | 1,295 | 3,570 | 1,745 | 6,610 |

Nomenclature:
A = Ground Test
B = System Ground Test
C = Flight Test
D = Manned Flight Test
E = First Operational Use
F = Spare

Notes:
1 Ten tests not required for man-rating because vehicle is modification of existing vehicle
2 No man-rating required

20-40% spare parts costs during initial evaluation tests is an average price

# SECTION VI

# PROBLEM AREAS AND RECOMMENDED PROGRAMS

## 6-1. PROBLEM AREAS

There are two pressing attitude-control problems peculiar to the orbital launch concept. These are

- Measuring and controlling the relative attitude of two vehicles in close proximity
- Attaching the astrotug to a second vehicle in such a manner that stable, efficient operation of the coupled pair is possible. The question of where and who applies what control torque is of particular importance from the standpoint of attitude control.

In addition, many problems not specifically arising from the orbital launch concept must be solved. The rate of the human, control accuracies, and dynamic response are typical example. A broad study of a few of the problems associated with development of attitude control systems for the OLO vehicles was attempted in the preceding sections, but much work remains to be done.

### a. The Relative Attitude Control Problem

There are three aspects of the relative attitude control problem: relative attitude of two vehicles must be sensed for control purposes, it must be maintained while

relative velocity impulses are applied, and it must be controlled to the requisite accuracy.

Many methods of sensing relative attitude can be envisioned, particularly when it is recognized that the important control parameter is not vehicle attitude, but rather the displacement of the nose and tail from the desired line of approach. Sensing techniques leading to more or less independent control of nose and tail position seem particularly attractive from the viewpoint of automation. Examples would include measurement of nose or tail displacement through hyperbolic phase comparison techniques, use of semi-independent beam-rider control of both the nose and the tail, or tugging on a rope attached to the vehicle while applying compensating thrusts to the tugged vehicle. Other sensors, such as radar, optical and infra-red devices, and direct visual observation should also be considered.

Maintaining the relative attitude of the two vehicles while correcting their relative velocity (or position) requires that the point on the vehicle whose velocity is to be controlled be defined. Restating the problem, it is necessary to select a pivot point when commanding a change in attitude with no change in position. The desired location of this pivot point will probably vary as the docking maneuver proceeds. It will not generally be at the C.G. In fact, the CG location will change with time and will vary from vehicle to vehicle. Again, some measure of independent control of the nose and tail seems desirable.

The method used to control relative attitude must also be established. Such factors as the degree of automation and the division of command authority between the two vehicles have an important bearing on operational success. As an example, maximum use of the crew is normally desired

for reliability and conservation of equipment weight, space and power. However, relative attitude control of two vehicles in close proximity is a function which seems particularly amenable to automation. Every docking operation will be the same. A high degree of accuracy but little judgment is required to control relative attitude. Finally, manual control would require extremely close attention for extended periods of time.

b. The Coupled Tug-Payload Attitude-Control Problem

Since the tug will probably not be employed in early orbital launch operations, this problem is not considered as urgent as the preceding one and initiation of work in this area may, if necessary, be delayed. Summarizing briefly, aspects of this problem include:

- Optimizing tug location with respect to the second vehicle
- Deciding between rigid and flexible couplers.
- Dividing the thrusting and attitude control torquing functions between the vehicles.
- Investigating the degree to which electrical interconnection of the two vehicles is desirable.
- Establishing the maneuverability, dynamic response, stability, and fuel efficiency of the coupled pair.

c. Attitude Control System Development Problems

Many stability and control areas require study and analysis in order to determine the optimum configurations consistent with maximum reliability and safety, and minimum cost and weight. Design studies must be initiated immediately

if the OLO vehicle attitude control systems are to be operational by 1966. Areas which should be studied as part of the system design study include:

- Establishment of stability and maneuverability requirements for the vehicle
- Investigation of the role of the crew. Primary, back-up, and emergency functions of the crew must be defined and the remaining functions automated
- Selection of firm system configuration
- Selection of sensors, control technique (pulse-width modulated vs on/off), reaction-torque generators, displays, and manual control equipments
- Investigation of vehicle stability and dynamic response to establish gain switching and shaping network requirements
- Allotment of error and unreliability margins to each subsystem and/or component
- Establishment of integration factors such as electrical, hydraulic, and pneumatic power; equipment weight and space; and the form, level, and impedance matching required for all signals flowing between subsystems. It is also necessary to establish which sensors may be common-use items with other systems and to define the interface equipment required for successful integration.

One particular problem worthy of special attention is that of developing bi-propellant attitude-control systems for the intermediate OLO vehicles. Extended performance and environmental testing of a representative bi-propellant

system would lead to increased confidence in system reliability. The state of the bi-propellant attitude control system art should then be comparable with that of cold gas systems, ensuring a trouble free qualification test program on the OLO systems.

## 6-2. PROGRAM RECOMMENDATIONS

Initiation of the following programs early in CY 1962 appears imperative if the OLO schedule is to be maintained:

### a. Relative Attitude Control Study

Tasks comprising this study are

- Comparison and evaluation of alternate methods of sensing relative attitude
- Synthesis of several tentative attitude/vernier velocity control systems and preliminary analytical comparison of them
- Analogue computer simulation of the several most promising systems followed by detailed computer evaluation. Manual control effectiveness would be investigated as part of this phase
- Preparation of system specification, including sensor selection, definition of a control technique, and component requirements.

### b. Attitude Control System Development

This task would include the following items for each vehicle control system considered:

- Investigation of the role of the man
- Selection of system configurations, sensors, attitude control torquers and control techniques.
- Analogue computer investigation of vehicle/control system dynamics
- Establishment of component specifications and interface requirements.

It is recommended that such a program be undertaken for development of control systems for the initial OLF, the Orbital Launch Vehicle, and the Unmanned Payload, in the near future. If at all possible, work should also be initiated on the storable bi-propellant development program and the coupled tug-payload attitude control study. Development of attitude controls for other vehicles can be delayed if absolutely necessary.

Finally, a discussion of a proposed reliability program, essential to the success of the recommended studies, has been included in the following section.

## 6-3. RELIABILITY PROGRAM

It is recognized that reliability is an indispensable requirement for successful Orbital Launch Operations. The only way to achieve high reliability is through an intensive, reliability-oriented design effort. This requires a developmental program in which

- Reliability is considered a primary design objective

  All personnel involved with the development program must be made aware that management has

established reliability as a major goal and is willing to pay the price required to reach this goal.

- Reliability must be treated as a quantitative design parameter
  Every set of alternate configurations or components should be compared on the basis of reliability. Every proposed system modification aimed at improved performance should be evaluated in terms of its effect on system reliability. Quantitative trade-offs are to be the basis for design decisions whenever possible.

- Reliability improving design techniques should be utilized whenever possible
  Several of the most important techniques to be used include: the use of standard, field-proven components, component derating, careful attention to packaging design in order to assure an environment suitable for the selected components, and a conscious striving for design simplicity.

- Reliability Control must be exerted positively through a Reliability-oriented design review.
  Every drawing is to be reviewed and given:

  A functional check - will it work?

  A component application check - are the components suitable for the application? Derating, environmental limits, and tolerance acceptability as to be emphasized.

  A packaging check - are the components mounted and located in such a manner that they will withstand the expected environment? Vibration

resistance and ability to dissipate heat must be given special attention.

A failure loading check - does failure of any one component impose undue loads on the rest of the system? Can the failure of a component be made non-critical so that continued degraded operation is possible?

This engineering reliability effort is to be followed by a thorough quality assurance program. A particular effort should be made to maintain very close liaison between Quality Control and Engineering in the early phases of the development cycle in order that Quality Control can inform Engineering of reliability-critical areas, and in order that Quality Control may become aware of potential difficulties.

# VOLUME V

## ORBITAL LAUNCH OPERATIONS

## PART III

## SECONDARY POWER

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# VOLUME III - SECONDARY (AUXILIARY) POWER

## 1.0 - SUMMARY

The Sperry Rand Corporation is supporting Marshall's Orbital Launch Operations Study (Vought Astronautics Contract NAS 8-853) by investigating the major Astrionic requirements for the program. The subject of this volume is Secondary (Auxiliary) Power Systems (APS), which covers the work accomplished by Vickers of Sperry Rand.

The results of studies conducted by Sperry Rand verify selection of the Snap 8 nuclear power plant for the OLF concurring with Vought and also establishes the advantages of using hydrogen-oxygen engines as an auxiliary power system for the OLF standby system as well as being the primary source of electrical energy for the OLV and the Astrotug. This cryogenic expansion engine has proved reliability through nearly 200 hours of testing. It has been designed and qualified expressly for space vehicle application where power is required ranging from two to thirty kilowatts with continuous operational time up to one week, and intermittent duty cycles up to one month (see Figure 3-1).

The $H_2$-$0_2$ chemical system recommended for the OLF, incoming payloads, Astrotug, and as a standby system for the nuclear powered OLF was selected on the basis of power-to-time requirements and its essentially 100 percent reliability, light weight, and low cost. Trade-off studies on the cryogenic auxiliary power versus fuel cells and solar cells have shown considerable substantiation of this choice.

Early models of the test engine achieved adiabatic efficiencies as high as 75 percent at operating pressure ratios ranging between 20 to 1 and 50 to 1. The brake specific propellant consumption (BSPC) achieved during these early tests was 1.3 pounds per horsepower-hour. From

further studies to be conducted in hydrogen-oxygen engines, efficiencies will approach 85 to 90 percent by 1964. Vickers has established this hydrogen-oxygen engine as an optimum auxiliary power unit for several applications in the orbital launch operation program.

The principal conclusions of this work are:

1. The auxiliary power aspects of the OLO program are within the time scale for state of the art development as conceived by Sperry Rand. No major technological breakthroughs are necessary to successfully establish OLO operations.
2. Certain follow-on study work is absolutely necessary to firmly establish plans for successful OLO operations. Since this work is within Vickers' capabilities, it is requested that assignment of it to Sperry Rand for the next funding period be considered. In the area of auxiliary power, this work includes:
   a) Establish the design parameters necessary for OLO operations so that system design can be initiated.
   b) Specifications for APS and life-science support must be prepared outlining complete requirements for the overall OLO program.
   c) A more penetrating study of the OLF, OLV, Astrotug, and incoming payload requirements must be made so that maximum performance and reliability can be achieved.

3. Certain follow-on work related to the OLF also appears necessary. This work is described as follows:

a) The shielding requirements for the Snap 8 power system appear to have a considerable influence on the overall configuration of the OLF. A more penetrating study of the shielding requirement and its effect on the arrangement and operation of the OLF is recommended.

b) The development of the runner-up OLF power plant utilizing solar energy is progressing at a fairly rapid pace. This progress should be continuously monitored to take advantage of any major new developments.

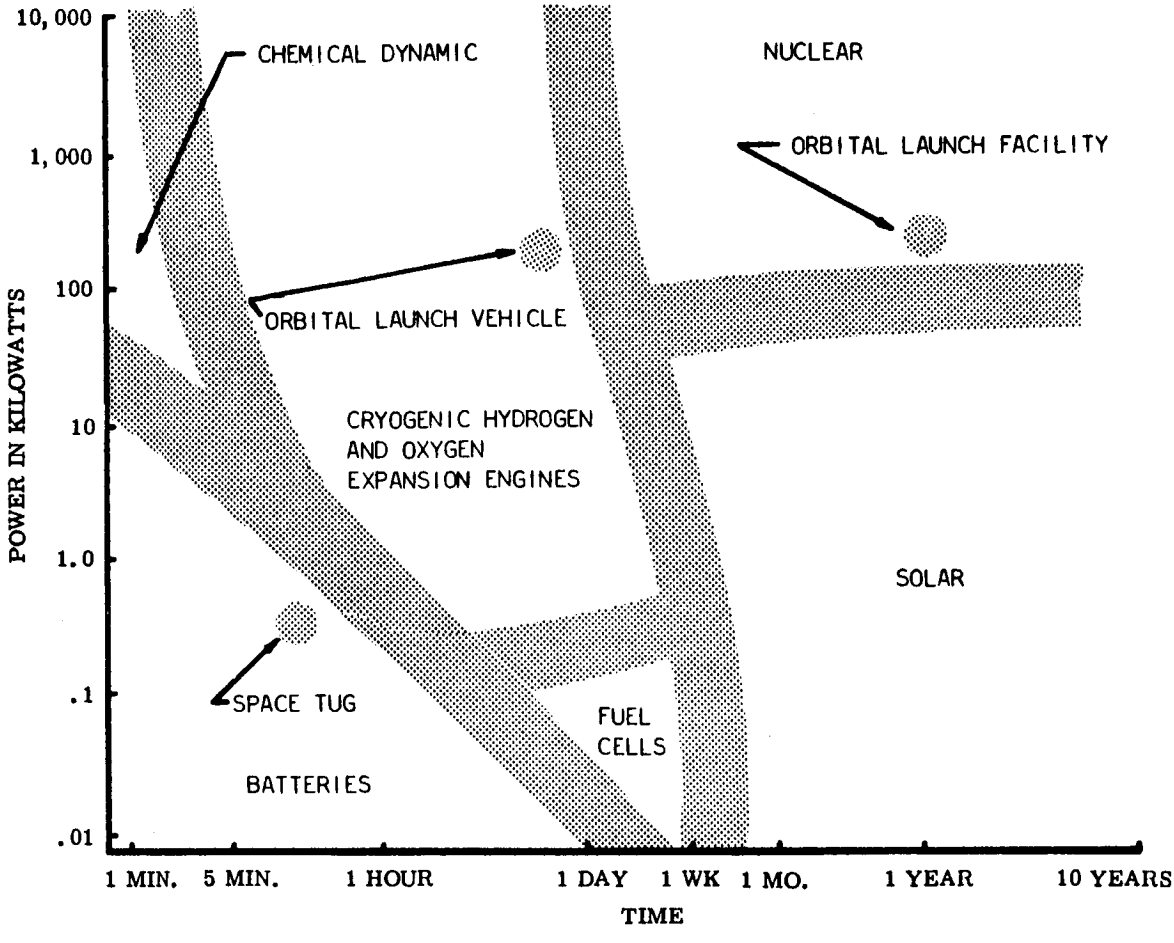

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POWER IN KILOWATTS
CHEMICAL DYNAMIC
NUCLEAR
ORBITAL LAUNCH FACILITY
ORBITAL LAUNCH VEHICLE
CRYOGENIC HYDROGEN
AND OXYGEN
EXPANSION ENGINES
SOLAR
SPACE TUG
FUEL
CELLS
BATTERIES
1 MIN.
5 MIN.
1 HOUR
1 DAY
1 WK
1 MO.
1 YEAR
10 YEARS
TIME

Fig. 3-1

## 2.0 - INTRODUCTION

The purpose of this report is to present the results of investigations conducted in the fields of auxiliary power systems. The Sperry Rand Corporation is conducting an across-the-board analysis of the various phases of Orbital Launch Astrionics to aid the Marshall Space Flight Center in developing a firm and well-documented plan for establishing the highly advantageous technique of Orbital Launch. The work is being conducted in connection with Vought Contract NAS 8-853, and is being guided by Vought work statements and coordinating documents. Every advantage is being taken of the results obtained by Vought in the earlier phases of the contract.

This phase of the Astrionic work is being funded by Sperry Rand as a contribution to establish the key engineering programs necessary to achieve success in this important endeavor.

Rapid technological advances in aircraft, missile, and orbital space vehicle programs have demanded concurrent progress in auxiliary power system (APS) concepts. Vickers Incorporated has assumed the responsibility of advancing this art by developing "packaged power" systems.

Space vehicle applications for auxiliary power systems dictate complete and independent power generating systems to replace the shaft power takeoff concept normally used on vehicles with reciprocating or turbine engines.

Since the forms of power required by space vehicles are similar

to those used by conventional vehicles (hydraulic, electrical, etc.), the auxiliary power systems evolved by Vickers are natural extensions of past product lines. Although many of these APS concepts are an integrated part of the prime mover power system, Vickers has also developed systems that are self-sufficient and extract primary power from an independent source.

Study the relative power requirements of various systems for the OLO program, the power required for each application and the possible need for multiple systems were primary factors considered. The studies investigated auxiliary power requirements aboard the Orbital Launch Facility (OLF), the Spacecraft (S/C), incoming payloads and Astrotug.

It became apparent early in the investigation of power plant requirements for the OLF, that the results of the Sperry analysis were closely following those obtained by Vought. The selection of the Snap 8 power plant system was confirmed on the basis of mission requirements, reliability, availability, size and weight. This is discussed further in Section 4.9 of this report. For standby power requirements for this facility and for power supplies for the other orbital vehicles, however, considerable further detailed analysis appears to be required.

## 3.0 - INITIAL CONSIDERATIONS

Orbital and Space-Flight Vehicles - The requirements for the on-board OLO power systems were taken from Vought Astro Design Data, Vol. II. (Power parameters were given for the four vehicles and are shown in table 3-1.)

The auxiliary power system is used primarily for attitude controls and electrical power, but will also offer a general power source. In applications where long-term nuclear power is desirable, an APS can be utilized as a standby power system.

Table 3-1 is a breakout of the vehicle power requirements for the guidance and control and overall electrical power requirements. Later in the report the discussion will be illustrated to show how the selected auxiliary power system will apply directly to these three vehicles.

TABLE 3-1 VEHICLES AND POWER SCHEDULE

| VEHICLE | POWER SYSTEM SCHEDULE | | |
|---|---|---|---|
| | YEAR | POWER REQUIRED | |
| O. L. F | 1968 | 30 | KW |
| | 1970-1975 | 60 | KW |
| SPACECRAFT | 1964-1966 | 1.5 | KW |
| | 1966-1969 | 2.5 | KW |
| | 1969-1975 | 3.5 | KW |
| | 3 MAN SPACECRAFT | | |
| | Guidance | 2.3 | KW/20min |
| | Misc. Power | 1.7 | KW/120 Hr. |
| | Lunar Rest | 1.4 | KW/x days |
| | Final Phase | 1.6 | KW/1 Hr. |
| | 6 MAN SPACECRAFT | | |
| | Launch | 3.2 | KW |
| | Mid Course | 2.8 | KW |
| | Lunar Land | 3.2 | KW |
| | Lunar Rest | 2.7 | KW |
| | Lunar Launch | 3.2 | KW |
| | Mid Course | 2.8 | KW |
| | Earth-Guidance | 2.5 | KW |
| ASTRO TUG | 1968-1975 | 1.80 | KW |
| | G. & C. | .45 | KW |
| | Lighting | .10 | KW |
| | Envr. & Inst. | .60 | KW |
| | App. Total Power (2 hour work period) | 1.2 | KW |
| INCOMING PAYLOAD | 1968 (Estimated) | 1.10 | KW |
| | G. & C. - Inst. | .2 | KW |
| | Attitude Control | .9 | KW |
| INTERPLANETARY TRAVEL | 1971 - 19xx | 300 | KW |

## 4.0 DESCRIPTION OF RECOMMENDED SYSTEM

Studies conducted by Vickers revealed that a dynamic hydrogen-oxygen auxiliary power system is capable of achieving performance equal to or better than that achieved with fuel cells. In addition, the chemical dynamic system can provide two additional desirable functions:

(1) operate on low pressure boil-off gases from the main propulsion system tanks,

(2) provide drinking water for the crew.

### 4.1 - Candidate System

A typical $H_2$-$0_2$ APS concept using the Vickers cryogenic engine is shown schematically in figure 3-1. In this system, an electrical alternator and a hydraulic pump are mounted on the $H_2$-$0_2$ engine, giving both mechanical and electrical power. Use of a through shaft in the alternator eliminates any need for gearboxes.

The hydraulic pump not only powers the vehicle servo system and positioning of solar devices but also operates in a bootstrap manner to power the hydrogen and oxygen propellant pumps.

The hydraulic fluid is cooled by circulating through a heat exchanger in the hydrogen line that leads to the engine combustor. The alternator is cooled by circulation around the alternator jacket and through a similar in-line heat exchanger. The hydrogen fuel also cools the electrical equipment compartment by absorbing heat from a nitrogen environmental control system.

The combustion supply pressures of hydrogen and oxygen are independently controlled by pressure regulating valves fed by reservoir pressures.

Since the hydrogen and oxygen propellants offer inherent engine system cooling, it is unnecessary to equip the engine with water jackets. Moreover, the engine exhaust is pure steam that can be condensed for crew drinking water. Water-payload penalties are thereby eliminated using the $H_2$-$0_2$ propellant concept.

Recognizing the importance of low propellant consumption and high reliability in power generating systems for manned orbital space vehicles, Vickers has instituted a research and development program demonstrating the feasibility of a positive displacement power plant for operation on propellant boil-off gases from the main propulsion system. A prototype illustration of this engine is shown in figure 3-2.

Advantages of such a power plant over fuel cells include:

1. Lower installed weight per kilowatt-hour.
2. Growth potential: lower installed weight per kilowatt-hour, lower brake specific propellant consumption (BSPC), and up to 1000-hour mission duration.
3. Fewer problems with zero-gravity conditions.
4. Capability of achieving higher thermal efficiency.
5. Early availability as a result of using proven concepts and hardware.
6. Low development cost due to parallel R & D programs now in progress.

The advantages of the $H_2$-$0_2$ system over conventional solar cells are:

1. The solar cell power system is presently much heavier than the combustion APU and it appears that by 1965 there will still

be a considerable weight advantage to the combustion unit.

2. Operational reliability of a solar array requires continuous orientation to the sun during the flight.
3. Protection from shock during the lunar landing and take-off operations is required. In particular, there is a real risk that the solar array could be damaged by a hard landing on the Moon's unknown surface.
4. Operational flexibility of the combustion APU is much greater because it is capable of operating in the absence of sunlight. For scientific purposes it permits investigation of lunar conditions during the lunar nighttime as well as daytime. Sunless operational capability provides an extra safety factor in case the OLV lands in deep dust, or the shadow of a tall mountain, or some other malfunction strands a crew on the return launch from the moon. Since the latter occurrence would probably take place toward the end of a daytime period the crew would face a problem in generating enough auxiliary power to maintain normal functioning of the vehicle internal environment and to communicate with a rescue vehicle. If the return launch rocket were also an $H_2$ $0_2$ system the crew could use the OLV fuel supply for survival. A rescue vehicle for this operation would probably operate during lunar nightime if the orginal exploration were a daytime expedition. Therefore

a combustion APU is mandatory! There is a measureable risk of failure to take-off from the moon and international prestige and political considerations make a successful rescue attempt very desireable as an illustration of comprehensive capability in the space field.

5. The solar cell system contains other difficulties which are probably solvable but still present considerable difficulties. Some of these problems are erection of large solar arrays, destructive effects of meteorites, and deteriorating effects of solar flares on solar cells.

Studies have been made on a regenetative power system concept, indicating the capability of achieving a BSPC of 0.85 lbs/hp-hr. Rapid availability of such a system is assured by utilizing proven thermodynamic concepts and power cycles differing from conventional power plants only

in the choice of fuel and oxidizer. Operation at conservative conditions of temperature, pressure, and speed further assure rapid availability of reliable components and systems.

A program is proposed which includes analysis and optimization studies, feasibility and performance testing of critical components, and similar testing of a complete "breadboard" auxiliary power system for an Orbital Launch Program.

Vickers is currently conducting a research and development program for Aeronautical Systems Division, Wright-Patterson Air Force Base, on a hydrogen-oxygen internal-combustion positive-displacement engine. An engine of the same or slightly modified design can be used in the OLO program. This current Air Force-sponsored program will also augment the OLO program in other major areas, including cylinder heat rejection, piston friction minimization, practical oxygen injection, and combustion properties of hydrogen and oxygen in internal combustion engines.

Plans are now in progress to develop a complete secondary power system that will provide environmental requirements for the crew on the OLO vehicles. By utilizing the steam exhaust and available nitrogen that would be required for cooling systems for electronic equipment, it would be possible to develop a temperature, pressure, and humidity control, and a breathing air source for the crew members. An integrated $H_2$-$O_2$-$N_2$ system could support the life sciences by providing both water and a source of ecological needs of the men. This type system would save many

pounds of overall weight if it were designed into a vehicle requiring approximately one month operational time.

By combining with the nitrogen the three natural atmospheric gases required to sustain life can be obtained. This system will not only provide the auxiliary power for any electrical requirements, but will provide the drinking water and a source of moisture for cabin humidity control. The superheated exhaust can provide a temperature control through a heat exchange process. The auxiliary power system can be utilized in a third manner; it could also provide an environmental pressure control because it can drive an air pressure pump for the cabins and crew living quarters. Basically, the $H_2$-$O_2$-$N_2$ system has the simplicity and coherent design that would be required for four other comparable systems. For these reasons, Vickers believes it worthwhile to recommend the system for the Vought OLO program.

1. Electrical power output - 2 kw
2. Generator efficiency - 90 percent
3. Hydrogen and oxygen both stored at 30 psia
4. Both hydrogen and oxygen available in required quantities as boil-off gases.

A regenerative-type power system has been analyzed for basic feasibility and is presented in the following paragraphs. It has been analyzed as though no heat sink would be available in space other than the incoming propellants. To make a fair comparison between fuel cells and

expansion engines, it is necessary to study a complete spectrum of systems operating with oxygen/hydrogen ratios between 2.0 and 8.0 (the latter ratio representing the oxygen/hydrogen ratio of a fuel cell). It may well be that there is an optimum combination of heat sinks and oxygen/hydrogen ratios which result in the most suitable power plant system. It is important to note that reciprocating power plant systems can be designed to operate at oxygen/hydrogen ratios of 8:1 with efficiencies of 60 percent or better; the resulting fuel consumption should be directly competitive with practical fuel cells and progressively competitive with solar cells and silver zinc batteries.

Even though an objective comparison between solar cells for 28-day operation and dynamic power plants is difficult without a detailed study, it is revealing to compare typical examples of these two types of systems. In general, weight and bulk of solar cells are very high. They are susceptible to meteoritic bombardment, cosmic radiation and are difficult to handle.

On the other hand, most chemical dynamic systems must employ methods of dissipating the exhaust. Condensors must separate the liquid from the gas under zero gravity conditions.

This problem is inherent in the tankage system. However, by using a bladder separating the working fluid from the inert gas it is possible to separate the cryogenic liquids from the pressurized gas. In general, the weight requirements for a hydrogen, oxygen system during a 28-day mission would be considerably less than solar cells. It must be kept in mind that

this system is proposed with the overall viewpoint of supporting life sciences. Table 3-2 shows the candidate systems for the four vehicles.

TABLE 3-2 AUXILIARY POWER SYSTEMS FOR THE OLO VEHICLES

| VEHICLE | RECOMMENDED AUXILIARY POWER SYSTEM |
|---|---|
| OLF | SNAP VIII<br>STANDBY $H_2$-$O_2$ SYSTEM |
| SPACECRAFT | $H_2$-$O_2$ APS<br>MANNED VEHICLE |
| ASTRO TUG | $H_2$-$O_2$ APS<br>MANNED VEHICLE |
| INCOMING PAYLOAD | 1. HOT GAS APS 1 kw for 1 hour<br>2. $H_2$-$O_2$ APS |

4.2 Assumptions

Although only a limited amount of information is available to formulate the bases for preliminary studies of possible auxiliary power systems, the following assumptions have been made to clarify the problem statement:

1. The auxiliary power system is designed as a self-contained unit.
2. Reliability is achieved by the use of redundant systems which weight and efficiency calculations take into consideration.
3. A high degree of mechanical design sophistication is considered permissible.

4. A progression of the state-of-the-art will result from any further study.

5. The influence of solar heating (which has been omitted from this preliminary study) can be made available.

6. Propellant tanks and motor lubrication systems will be designed to operate in a zero-g environment.

7. APU operation may be monitored by personnel and/or remote servo systems, and the power units may be completely regulated during flight.

8. No special "overload" components are required and power over the full range is supplied by the single system.

4.3 - Parametric Studies

During the research and development phase of the internal combustion engine program, a continuing study is being made of the important parameters in an effort to minimize development time.

1. Weight and Volume - Materials, packing techniques, and hardware design will be investigated for minimum weight and space requirements.

2. Power Requirements - Sufficient growth potential will be incorporated into all candidate systems to allow for future growth. The most effective means of insuring the desired growth potential will be studied.

3. Availability and Development - For an internal combustion positive-displacement engine to be competitive, it must be able to meet any required availability dates. Proposing two engines of different levels of development allows considerable flexibility in design schedules and ensures that the power system will remain competitive for many years.

4. Reliability - The low fixed weight and stable construction materials inherent in an internal combustion positive displacement APU give it an advantage over radiation and temperature sensitive competitors such as fuel cells and batteries.

5. Maintainability - Maintenance and servicing capability will be considered as basic design goals.

4.4 System Parameter Sensitivity

The basic design parameters may be changed or modified as the problem becomes better defined and more information becomes available. The sensitivity of the basic APU to any change in requirement is a major factor to consider in the selection of an auxiliary power system. A discussion of the effects of requirement and state-of-the-art changes on the $H_2$-$O_2$ I. C. engine APU follows:

1. Weight and Volume - The weight of the system is extremely flexible, and a weight reduction can always be effected at the expense of some design complexity. Volume requirements are as low as any competitive system and the promise of future efficiency improvements can lower fuel weight and volume.

2. State-of-the-art Development - A major breakthrough in the fields of batteries, fuel cells, and other direct conversion devices will be necessary before the relative competitive position of the positive displacement internal combustion engine is changed. It is not possible to predict such a breakthrough or to anticipate its results although continuing refinements in all systems are anticipated, the Hydrox Internal Combustion engine is expected to remain a major candidate.

4.5 S/C Auxiliary Power Unit Description

The requirements for an auxiliary power unit for the 3-man Spacecraft are given in table 3-1. Enough growth potential is incorporated to permit the use of this system in a 6-man vehicle with minor modifications. Two engine design levels are discussed: (1) a simple engine that will be available in 1963 and (2) a more advanced model, benefiting from stoichiometric combustion, that will be available in 1965. Both systems are compared with a fuel cell system and with a solar cell system. All energy requirements are specified except for the time spent at rest on the surface of the moon. To allow flexibility, propellant and tankage weights are expressed as a function of the lunar rest time. The following equation gives electrical energy available from the system.

gives electrical energy available from the system.

$$E = 247.3 + 33.6t \quad \text{kilowatt-hours,}$$

where t = number of days at lunar rest

Graphically, this equation can illustrate the electrical energy and weights as a function of lunar rest time in days as shown in figures 3-3 and 3-4. The

support calculations for these weights are shown in the systems breakout chart (table 3-3).

TABLE 3-3 OLV SYSTEMS CALCULATIONS

| | Advanced I.C. engine | Simple I.C. engine | Fuel Cell |
|---|---|---|---|
| Oxygen | 210+28.4t | 170+23.2t | 216+29.3t |
| Oxygen tank | 42+5.7t | 34+4.64t | 43.3+5.9t |
| Hydrogen | 26.2+3.6t | 113.5+15.5t | 27.0+3.7t |
| Hydrogen tank | 21.0+2.8t | 90.8+12.4t | 21.7+2.9t |
| Total Propellant | 299.2+40.5t lbs | 408.3+55.74t lbs | 308.0+41.8t lbs |

Adding the above weights to the engine hardware,

| | | | |
|---|---|---|---|
| Power Units (3) | 24 | 18 | 216 |
| Alternators | 45 | 45 | --- |
| Pumps | 3 | 3 | 3 |
| Plumbing | 10 | 10 | 10 |
| Total | 381.2+40.5t lbs | 484.3+55.74t lbs | 537.0+41.8t lbs |

t = number of days spent at lunar rest.

Solar cell weight is composed of:

| | |
|---|---|
| Cells and structure | 280 lb |
| Glass | 245 lb |
| Storage | 325 lb |
| | 850 lb |

As is shown in figure 3-4, for lunar stay times of less than one week, the simple $H_2O_2$ engine and fuel cells are comparable to each other with solar cells gaining the ascendancy for periods from one to two weeks. At the end of two weeks max, the solar cell becomes inoperative for the two week lunar night.

The advanced $H_2O_2$ engine is superior to all three above energy sources from the weight standpoint for any length of lunar stay.

The solar cell weight given in figure 3-4 has assumed a 30% decrease in weight over 1962 state-of-the-art solar cell development. Estimates for weight decrease run as high as 55% by 1965. The 30% decrease is considered a reasonable reduction factor for this report. The battery weight given is not sufficient to maintain the required KW output for earth orbital operations but is adequate for temporary shut downs required for mid-course and lunar terminal maneuvers.

Efficiency - A possible brake specific propellant consumption of 0.60 lb/hp-hr is assumed for the most efficient operating level of the advanced internal combustion hydrogen-oxygen engine. This number is based on a stoichiometric combustion ratio ($O_2/H_2$ = 8/1). For the simple internal com-

bustion engine, a BSPC of 0.74 lb/hp-hr at a mixture ratio of $O_2/H_2$ = 1.5/1 is assumed. The minimum BSPC of the fuel cell system is 0.65 lb/hp-hr at stoichiometric mixing. In figure 3-5, the off-design point efficiency of these three systems is shown. It is seen that overload operation on one engine will raise the BSPC of the advanced and simple IC engines to 0.63 lb/hp-hr and 0.77 lb/hp-hr, respectively. No correction was made for fuel cell failure since fuel cells were assumed to operate singly at the optimum load.

The overall thermal efficiency of the $H_2$-$O_2$ system is over 70 percent which is considerably higher than fuel cell efficiency.

An electrical conversion efficiency of 90 percent was assumed for all systems. This represents an alternator efficiency in the reciprocating engine systems and a DC-AC inverter efficiency for the fuel cell.

Tankage - The total propellant weight was calculated from the energy requirement given by the equation for electrical energy, using the applicable BSPC. Tankage weight was then calculated by using the following assumptions:

Oxygen tankage weight = 20% oxygen weight

Hydrogen tankage weight = 80% hydrogen weight

Tank weight estimates are based on the results of Vickers studies on advanced concepts in zero-g cryogenic tankage for space missions. See reference 6 for a discussion of the rotating fluid tank concept.

Motor Weight - Motor weight was calculated for the two reciprocating engine systems as follows:

Motor BMEP of 200 psi at maximum rpm was assumed. The greatest power output necessary is 3.6 hp (one motor, generating 2.4 kw @ 90% efficiency). If maximum engine speed is 6000 rpm for this extreme condition, an engine displacement of 1.2 $in.^3$ results.

Extrapolation from the weights of small 1-cylinder industrial engines in this displacement range yields a weight of 7.6 pounds. A conservative assumption of 8 lbs/motor was used. This weight includes engine speed controls and accessories.

OLV Reliability Discussion - Manned space flight demands a high level of reliability that can be achieved by the use of redundant systems. The following basic rules, taken from reference 2, were used to determine the necessary redundancy:

1. An orbital launch vehicle reliability of 0.85 requires an auxiliary power system reliability greater than 0.9999.

2. Separate, complete, and fully installed power units are desirable. The number of units depends on the reliability of each.

Well documented reliability figures for mechanical systems operating in a space environment will not be available for years. If a single unit reliability of 0.95 can be attained by the proposed internal combustion hydrogen-oxygen engine, the system reliability for three parallel units is .999875. This number represents the probability that all three units will not fail during the mission. In actual practice, two units will be operating under reduced load at all times, thus increasing the dependability of an individual unit. Also, the unit (or units) not operating will not deteriorate

with time for the periods considered in this mission. It is believed that these considerations will permit a three-engine APU to attain the desired reliability.

The one-engine reliability figure of 0.95 is an estimate of turbine reliability under space conditions. This reliability may be easily attained or surpassed by a small reciprocating engine since the experience of seventy years of design technology may be brought to bear on all problems which arise. The reliability of the engines used in the OLV APU depends primarily on effectively incorporating the results of the development effort and environmental testing effort into the final design.

An estimated fuel cell system reliability of 0.96 is estimated in the literature. Three parallel systems are needed to surpass the required reliability level. Due to the high fuel cell system weight, this requirement imposes a more severe penalty on the fuel cell APU than it does on the internal combustion engine APU.

Maintainability of OLV and Astro Tug - When human beings are available in a space system a great deal of complexity and system redundancy may be eliminated by effectively incorporating them into the system. Conversely, the elaborate environmental controls needed to sustain human life can be made more efficient if waste heat and by products are effectively re-cycled.

There is no reason why a standby power system for the orbital launch facility could not be designed to be repaired, serviced, and dismantled by personnel on board. Spare parts and required tools could be made available

to the facility.

It would be possible to relax the reliability requirements of the assembled unit if it could be easily serviced and overhauled by men of moderate skill working under adverse conditions. The success of non-professional automobile mechanics over the last fifty years attests to the maintainability of an internal combustion engine under these conditions (a contention that cannot be held for fuel cells or turbines).

Space tug APU's could also be made repairable and easily replaceable or exchangeable. The weight savings offered by an internal combustion engine system over a battery system for long operative periods depends on reducing the fuel weight, spare parts, extra motors, and tool weights for the internal combustion engine to less than the battery weight plus the weight of fuel used to charge the batteries.

Human decision-making with respect to instrumentation could also eliminate complexity. All machine decision-making is inherently wasteful and excessively redundant since it must be reduced to go-no-go situations, precluding judgment.

A discussion of alternate systems in which human and sealed-cabin waste heat is effectively re-cycled into the system is presented in the following section. It must be remembered that a fuel cell or battery system cannot make the effective use of this waste heat that is possible with a cryogenic expansion engine.

4.6 - Alternate Systems for OLV

The following alternate approaches to the problem of space electrical

power generation are being considered by Vickers. The feasibility of each approach cannot be determined by the information presently available on system requirements and space operating conditions. Each approach may prove valuable as a means of effectively integrating the APU into the overall vehicle.

Alternate System No. 1 - Main Tank Propellant Boiloff on OLV - An auxiliary power system which uses boiloff gases from the main propellant tanks provides weight reduction in the following areas:

1. Total tankage
2. Total hydrogen under storage
3. Main hydrogen tank insulation (since a higher boiloff rate is permissible)

A complete system design incorporating the use of boil-off gases would require detailed knowledge of the entire vehicle heat balance and the trade-offs between insulation weight and propellant weight. Even if boil-off rates are not a critical problem it may be desirable to utilize the main tanks in the interests of efficiency.

An auxiliary power system weight comparison considering fixed weight alone is shown below. This weight, taken from the OLV systems calculation, is independent of mission duration and illustrates the low fixed-weight advantage of internal combustion engine system.

| | |
|---|---|
| Advanced IC engine | 82 lbs |
| Simple IC engine | 76 lbs |
| Fuel Cell | 229 lbs |

Alternate System No. 2 - Personnel Cabin Heat Exchange on OLV - Vehicle cooling was not included in the OLV system description because available heat from the crew compartment and equipment has not been clearly defined. For an accurate study of an integrated system, the following values should be specified.

1. Heat rejection of crew to the vehicle.
2. Heat available to the APU from the vehicle.
3. APU output power which is returnable to the APU through an APU-crew cabin feedback loop.

These values are combined into an overall heat feedback ratio which is usually represented as a percentage of APU output power. Heat feedback can be removed by radiation or by use of the APU hydrogen supply as a heat sink (or both), depending on mission duration, available feedback, and mission requirements.

For most missions that Vickers has studied in the past, the heat capacity of the hydrogen supply for an APU similar to the one described is usually less than the total available heat feedback. Therefore, a radiator may be necessary for vehicle cooling. Another method is heat feedback to the main propellant tanks (particularly if boil-off is being used). A drawing of an APU incorporating heat feedback is shown in figure 3-6.

Alternate System No. 3 - The Use of Auxiliary Power System Exhaust as a Water Source - Water in a manned spacecraft is needed for two applications:

1. Crew drinking water
2. Re-entry cooling

Considerable water will be recovered from crew waste. If more is needed, condensation of the APU exhaust gases may reduce vehicle weight. Figure 3-7 shows the internal combustion regenerative power system in schematic form. Using the present Vickers research engine in this system, approximately 75% of the vehicle exhaust water could be recovered.

Alternate System No. 4 - Standby Power - Orbital Launch Facility - Because the reliability in space is in a state of indecision, the feasibility of a chemical standby power system for the nuclear APU of the 36-man orbital launch facility cannot yet be determined. Assuming that this concept is desirable, some representative system weights have been calculated and are presented below for a system of 48-hour duration. As a comparison, a nuclear APU would weigh approximately 20,000 pounds (two units).

| | |
|---|---|
| Oxygen fuel and tankage | 2200 lbs |
| Hydrogen fuel and tankage | 2200 lbs |
| Engine weight (60 in.$^3$ displacement, 90 hp) | 80 lbs |
| Alternator weight | 120 lbs |
| Plumbing | 25 lbs |
| Total | 4625 lbs |

A similar engine using storable bipropellants may be necessary if long-term availability is necessary (1-5 years storage). Some initial fuel weight would be sacrificed for storability.

4.7 Astro Tug APS - Two alternate auxiliary power systems for the space tug are compared below. A small cryogenic $H_2$-$O_2$ internal combustion system with two engines and fuel for two round trips is compared to a moderate-

discharge-rate secondary battery system using two silver-zinc battery units. It is believed that either system will satisfy the power and reliability requirements for the space tug auxiliary power system.

| | Battery | Reciprocating Engine |
|---|---|---|
| Number of redundant systems | 2 | 2 |
| Conversion efficiency | 90% | 90% |
| Electrical power level | 1.2kw | 1.2kw |
| Duration | 2 hrs | 2 hrs |
| Fuel weight | ----- | 6 lbs |
| Tankage weight | ----- | 5.7 lbs |
| Fixed weight | 100 lbs | 32 lbs |
| Total Weight | 100 lbs | 43.7 lbs |

The power requirements of the Astro Tug were shown in figure 9-5 of the Vought Design Data, Vol. II. A two-hour work duration is required with a six-hour battery charge. If a hydrogen-oxygen system were used it would offer a substantial decrease in charge time. The six hours would be reduced to a matter of minutes to refuel the six pounds of fuel weight required for vehicle operation. Slightly more than this would be required for a manned vehicle.

Quick-disconnect couplings, automatically operated on the front of the Astro Tug and mounted externally on the platform, would allow an exchange of required fuel of hydrogen and oxygen.

4.8 - Incoming Payload APS - According to recent information on the OLO program there is a distinction between retrieval and rendezvous of pay-

load. The retrieval phase of an incoming payload operation will take place within 500 feet of the platform or a designated position. However, rendezvous will be considered as the long range operation beginning at earth launch and ending at the retrieval phase.

On the basis of this we have calculated the requirements of an auxiliary power system to provide one kilowatt of power for attitude controls and generator power for electric or electronic equipment and related components. It is estimated that this one kilowatt is approximately 800 watts of electric power and 200 watts of mechanical power or hydraulic power. The trade-off studies for attitude control show that 14 nozzles would be required for complete directional and attitude control for the incoming payload. With the high reliability only 200 watts will be required for this nozzle system. The remaining 800 watts will be used for electronic equipment power.

As this is only a one-shot affair, it would be possible to propse either of two systems. The first is a hot gas system whose weight is extremely low and which would offer a required power for one hour's time. Secondly, if the payload were manned, it would be possible to incorporate a hydrogen-oxygen engine.

## 4.9 OLF SECONDARY POWER

### Basis of Consideration

To provide a base line with which to measure the performance and weight of various secondary power systems, 30 KW and 60 KW power output levels have been assumed. These values are convenient in that the Snap 8 nuclear power plant is being specifically designed for the above two power outputs. Other power sources can readily be scaled to these values whereas modifying the Snap 8 outputs would require extensive investigation into the basic design of the Snap 8 with attendant confusion.

### 4.9.1 Snap 8

The Snap 8 power supply is under development by the AEC and NASA. This power plant for 30 KW and 60 KW operation weighs 1500 lbs and 2500 lbs, not including the shielding required for personnel protection, and employs 7.0 kg fuel to produce 30 KW with a probable life of one year. The shielding required varies with the distance between the OLF and the Snap 8 supply, the amount of time crew members will remain in orbit, and which sets of figures are to be believed as being the maximum radiation dose which humans can withstand and not suffer severe permanent damage. This shielding estimate varies from 10,000 lb for spot shielding to 36,000 lb for hemispheric shielding. For purposes of comparison with other APU's, the more conservative weight of 36,000 lb for the total Snap 8 weight will be assumed.

This shielding is an appreciable fraction of the total mass of the orbital launch facility. Because the crew will spend a considerable porportion of their time outside the facility, adequate protection from the nuclear reactor can be obtained only if most of the reactor shielding is located around the powerplant, which is separated from the facility, rather than around the facility itself. Unfortunately, this precludes employment of the reactor shielding in double service; as a shield against both radiation from the nuclear reactor and radiation from the solar bursts. Reactor shielding weight does not serve any additional purpose that mitigates its undesirability. Therefore, trade-off studies would give great advantage to a very small shielding weight.

Operation of the Orbital Launch Facility requires regular rendezvous with supply vehicles carrying both men and equipment. The close proximity of the nuclear powerplant to the facility (approximately 50 feet) and the dependence upon maintaining this distance from the reactor creates a hazard during the final stages of rendezvous. A larger separation of the reactor from the facility is desirable to simplify the rendezvous problem.

Future orbital space flights may show that man cannot function effectively in a weightless environment for extended periods of time greater than approximately several days. This circumstance would require the generation of artificial gravity by rotating the facility to generate a centrifugal force. Other experiments on men have shown that prolonged rotation at rates greater than 1.7 RPM produce nausea by disturbing human metabolism through the balancing mechanism in the middle ear. Therefore a space facility

designed to generate a "safe" artificial gravity rotates at approximately 1.0 RPM on a radius of approximately 6,000 feet. If the biological nature of man proves to be such that weightlessness must be avoided, then the facility design problems would be greatly aggravated. However, it is possible to take advantage of a centrifugal configuration to greatly reduce the weight of the shielding required for the nuclear powerplant and reduce the radiation hazard involved in rendezvous operations with the facility. Figure 3-8A illustrates the type of design that might be considered to solve the above problems in an advantageous manner.

The technique of Figure 3-8A is only one possible configuration which might be desirable under specific circumstances. The entire field of shielding weight, structural arrangement, rendezvous and safety offers both opportunities and requirements for extensive investigation. This is a complex of problems which may require considerable effort in future studies.

Shielding requirements have a great effect on weight in orbit, but the basis for selection is not well defined at present. Therefore a study in depth to determine merits of various facility arrangements and different shielding is recommended.

4.9.2 CHEMICAL SECONDARY POWER

A highly efficient 30 KW chemical system consuming only 1.0 pound of fuel per KW - HR. will in a year's time burn a weight of fuel which is many times greater than the weight of the nuclear reactor plus its shielding.

Wt. of Fuel = 30 KW x 365-1/4 days x 24 hrs/day x 1.0 fuel/KW-HR.

Wt. of Fuel = 263,000 pounds for one year.

This weight is enormously greater than approximately 36,000 pounds for the weight of the Snap 8 nuclear power plant plus shielding.

4.9.3 SOLAR POWER SOURCE

There is a need for an alternate type of power plant in case it becomes very desirable to eliminate the nuclear unit from the design. Possible substitutes are various solar power plants described below:

Photovoltaic Power Systems use semiconductor solar cells which absorb light quanta and convert it to electricity within a single crystalline cell.

Photoemissive Solar Converters use large photo sensitive surfaces which eject electrons to be captured on adjacent screen electrodes. They employ some of the technology of both photomultiplier and image intensifier tubes.

Thermoelectric Solar Converters employ a semiconductor thermoelectric material to generate power from temperature difference between a pair of specially coated surfaces, one of which is heated by the sun.

Solar Dynamic Systems are Rankine cycle turboelectric generators using the sun as a heat source, radiation to space as a heat sink, and using exotic materials to secure high performance under difficult conditions.

Solar Thermionic Systems generate thermionic diode power when the cathode is heated by sun's rays from a solar concentrator. Vacuum type diodes are now on the market and cesium vapor diodes with higher efficiency will soon be available.

All high power, high performance solar power plants have large reflectors called solar concentrators whose purpose is to concentrate the sun's rays on the solar energy conversion equipment. A 30 KW installation requires a large unwieldy solar collector on the order of 64.0 feet in diameter. In addition, the collector must be aimed at the sun with an accuracy ranging from several degrees down to approximately 0.1 degree depending on the choice of system. Photovoltaic, photoemissive, and thermoelectric systems require several degrees accuracy, solar dynamic systems require approximately 1.0 degree accuracy and solar thermionic converters require the maximum accuracy approaching 0.1 degree. Most of the weight of a solar static system designed for satellite service around the earth is in the storage batteries which supply power while the facility is orbiting through the shadow

of the Earth. Data on the component weights of a solar cell power plant orbiting about the Earth has been obtained from WADD Technical Report 60-699, Energy Conversion Systems Reference Handbook, Volume IX - Solar System Design by W. R. Menetrey. Figure 3-8B from the report shows the weight of the component parts in pounds/KW versus satellite altitude for different satellite altitudes with the equipment available in 1962. At the Orbital Launch Facility, altitude of 300-400 miles, system weight is a maximum. Figure 3-8C extends the above data to the future with total weight curves for both 1965 and 1970. The startling drop in total system weight from 800 pounds/KW in 1962 to 150 pounds/KW in 1970 is a projection based primarily on reduction in battery weight. This is expected to occur either as a result of the use of new types of batteries such as fuel cells or improvements in reliability in presently known batteries such as silver-zinc or nickel-cadmium batteries. Present reliability of nickel cadmium batteries limits the depth of discharge to 10.0% of maximum for the repeated cycling application characteristic of relatively low altitude Earth satellites. Therefore improvements in battery reliability without any increase in maximum battery charge could produce the future performance increases projected in Figure 3-8B. This information on battery development applies to all solar static power plants in low satellite orbits where the storage batteries are the primary component weight in the system. Note that the above weights for a solar cell system gives a total weight for a 30 KW solar cell system of 24,000 pounds by 1962, 4500 pounds by 1965, and 2100 pounds by 1970. However, it is questionable whether the above weights per KW can be

maintained at a power level as high as 30 KW for a solar cell system which are inherently suited to lower power levels. But, if the projected development of these systems takes place over the next few years then they would provide a power plant appreciably lighter than Snap 8 with its shielding.

The solar dynamic system appears to be most suitable for the power level of very approximately 4 KW to 20 KW. Proper comparisons for this type of power plant can be obtained only by employing complete designs. An excellent example of this type of solar power plant is the 15 KW Solar-Mechanical unit which WADD is now funding. This is a 15 KW solar power plant weighing 819 pounds and using a 40.5 ft. diameter solar concentrator. It employs several different chemicals for thermal storage when the satellite is in the Earth's shadow. To compare this with other techniques of power generation it is most convenient to employ weight per unit power. This 15 KW unit weights 55 pounds per KW which is the lowest of any system investigated in this report. Aside from the large solar concentrator, this unit has the disadvantages that it will not be ready for approximately four to six years if there are no major delays, and the power output is 15 KW instead of the 30 KW required by the Orbital Launch Facility. There are three alternatives to the latter problem:

1. Use two 15 KW units
2. Attempt to purchase a 30 KW unit with a considerable increase in lead time.

3. The 30 KW power requirement, which is only an estimate, could be reduced to 15 KW supplied by a single solar dynamic unit weighing 819 pounds. It has been estimated by the cognizant personnel at WADD that a 50 KW unit would weight about 45 to 50 pounds per KW. In addition, they have estimated that this specific weight is a minimum at this power level and would rise for either an appreciably higher or lower power level.

Comparative Reliability of Proposed Power Plants.

Although the Snap 8 nuclear power plant is intended to have an inherent reliability capable of insuring at least one year of operation, it does possess the disadvantage that it must be shut down for all maintenance and even then it must be handled with manipulators through a protective shield. Any reliability of power continuity during maintenance periods almost certainly requires the use of at least two Snap 8 power plants with the increased weight. Solar cells may suffer slow deterioration over a long period of time caused by bombardment by X-rays and gamma rays particularly during severe solar flares. However, this deterioration is sufficiently slow such that over a one year period the reliability should remain quite high. Because the solar cell power plant uses many solar cells, whose weight is not a major portion of total system weight, it is possible to greatly improve reliability in the solar cell bank by wiring them in a mixed series parallel arrangement having a great increase in redundancy.

The storage battery portion of the system can be easily separated into many small units with sufficient switching equipment to remove a defective battery from the circuit. The static electrical conversion and regulation equipment possesses more than adequate life. In addition, the entire system is readily repaired by the crew without any special precautions.

The solar dynamic system is a mechanical system constructed from materials that are not deteriorated by space radiation. Again, the system is readily repaired by the crew without any special precautions against radiation. Two or three 15 KW power units would permit one unit to be shut down for repairs while the other unit(s) provided continuity of electrical power.

4.9.4 SELECTED SYSTEM

The Snap 8 system is the only system which has a reasonable development schedule for dovetailing into the overall OLO schedule despite its disadvantages. These disadvantages are that presently nuclear power plants are relatively difficult to maintain; present a political problem in their operation; require heavy shielding, and have what is perhaps an unnecessary radiation hazard to the facility crew. Although other power plants such as the 15 KW solar dynamic system appear vastly more attractive from the weight and size standpoint than either solar cell or nuclear, the very severe limitation of 15 KW imposes stringent limitations on the OLF equipment. The use of two or three solar dynamic systems is feasible but requires an operational analysis of the use of the volume immediately surrounding the OLF due to the large solar concentrator areas required. Sperry Rand

proposes to study the details of Orbital Launch Facility operation to determine whether a large collector would interfere with the proper functioning of the facility. The study would determine what changes are necessary in facility operation to accommodate the operation of the solar power system. A comparison will be made to determine the net change in the usefulness and cost of the facility for the new configuration.

Solar Power plants are presently behind nuclear power plants in larger power ouputs, require large solar collectors which must be oriented to the sun, and will probably never be suitable for very large power outputs. At the present time the solar power art is developing rapidly requiring careful observation to provide the latest information on developments in this field. We therefore recommend as a specific follow-on job that Sperry Rand will visit various laboratories where the latest techniques are being developed to provide a systematic up to date comparative summary of the state of solar power plant development. For the higher power levels required by the Orbital Launch Facility, solar power plants are in the early planning or development stages. Their projected performance and reliability five or more years from now contains a measurable element of chance. However, in several years solar power development may improve to the point where it will possess a high probability of success. This information is necessary for any fundamental decisions involving the choice of secondary power sources on the Orbital Launch Facility.

## 5.0 - COST AND SCHEDULE

The S/C, OLF standby, payload, and astrotug power system development plan and schedules which follow indicate the magnitude of effort required before the proposed power system can be expected to achieve operational status. This program is a natural extension of the present Vickers work on an Air Force contract for development of an internal combustion hydrogen-oxygen engine for space power application. The present contract work allows the projected program schedule to be shortened correspondingly.

The R & D phase of the engineering development program can be subdivided into three areas. These are: 1.) Evaluate current Air Force contract work at Vickers to demonstrate feasibility of the internal combustion hydrox engine concept and obtain basic data; 2.) Demonstrate feasibility of the regenerative power system concept; 3.) Demonstrate feasibility and develop it to production prototype status of the advanced stoichiometric chemical power system. Beyond this R & D phase is an additional effort required in product engineering, service, reliability, qualification, logistics program support, etc., to achieve true operational status. The total program is outlined below (see figure 3-8):

I. R & D Phase

A. Current Air Force contract work.

1. Demonstrate feasibility of hydrox internal combustion engine concept.
2. Obtain basic data on combustion, cooling requirements, etc.

3. Determine performance at specific conditions.

B. Basic power system development (system R & D work leading to feasibility demonstration of complete "breadboard" power system operating at fuel-rich mixture ratio)

1. Perform power system studies, thermodynamic analysis, and controls analysis; predict system performance; identify probable problem areas.

2. Test "breadboard" system; determine actual system performance; obtain basic data on system performance, stability, and general characteristics; resolve basic problem areas.

3. Demonstrate feasibility of complete regenerative power system operating on cryogenic propellants.

C. Advanced power system development (system R & D work leading to production prototype status of complete power system to operate at stoichiometric mixture ratio)

1. Perform engine and power system studies, thermodynamic analysis, and controls analysis; predict system performance; identify probable problem areas.

2. Test "breadboard" system; determine actual system performance; obtain basic data on combustion characteristics, system stability, and general operational characteristics; verify and resolve basic problem areas.

3. Demonstrate feasibility of complete power system operating on cryogenic propellants at stoichiometric ratio.

4. Develop system to production prototype status; resolve prototype system problems.

II. Operational Qualification Phase

A. Operational system development

1. System design

2. Development testing

B. Reliability program

C. Production engineering

D. Service engineering

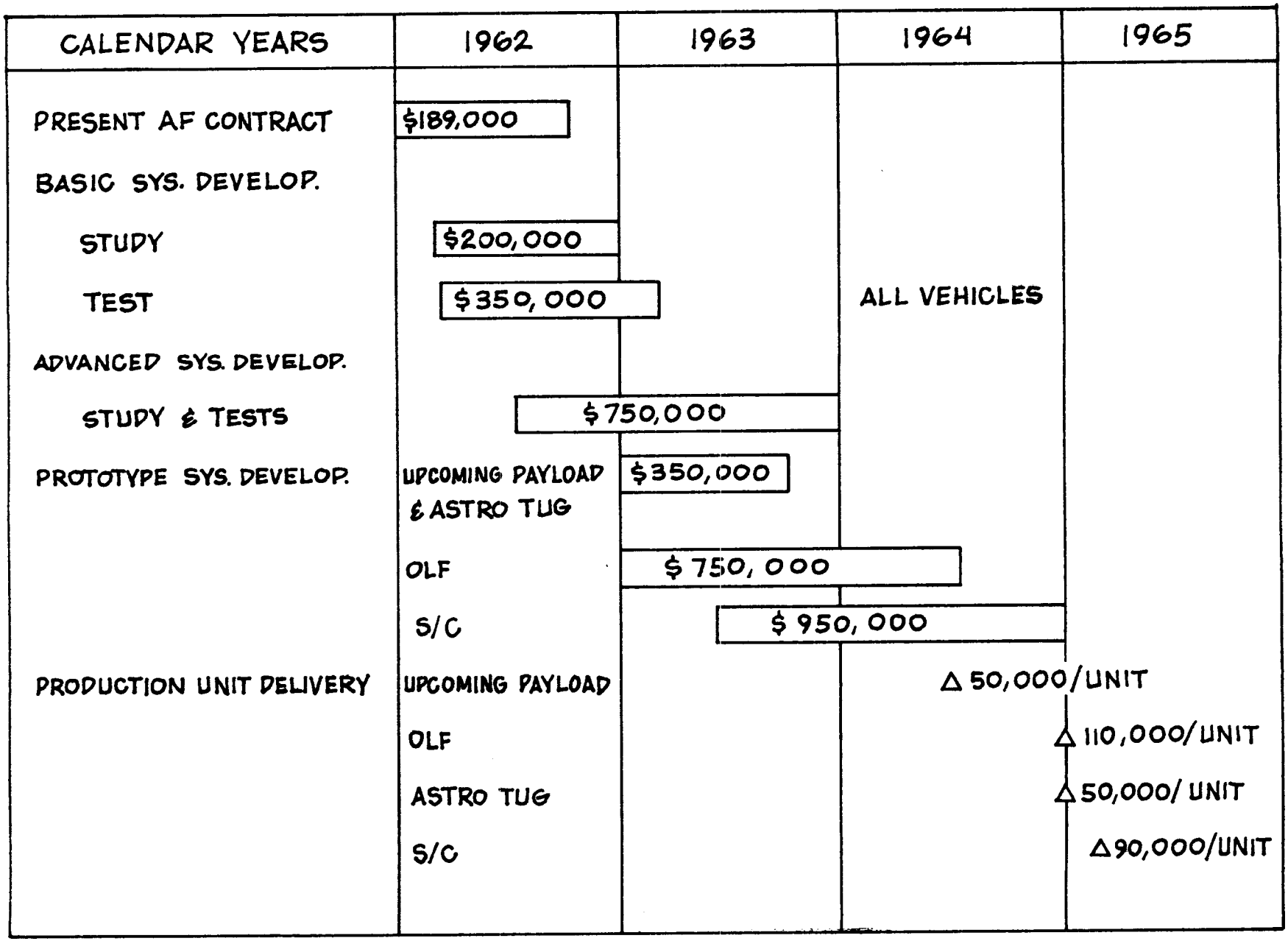
E. Manufacturing

F. Logistics planning

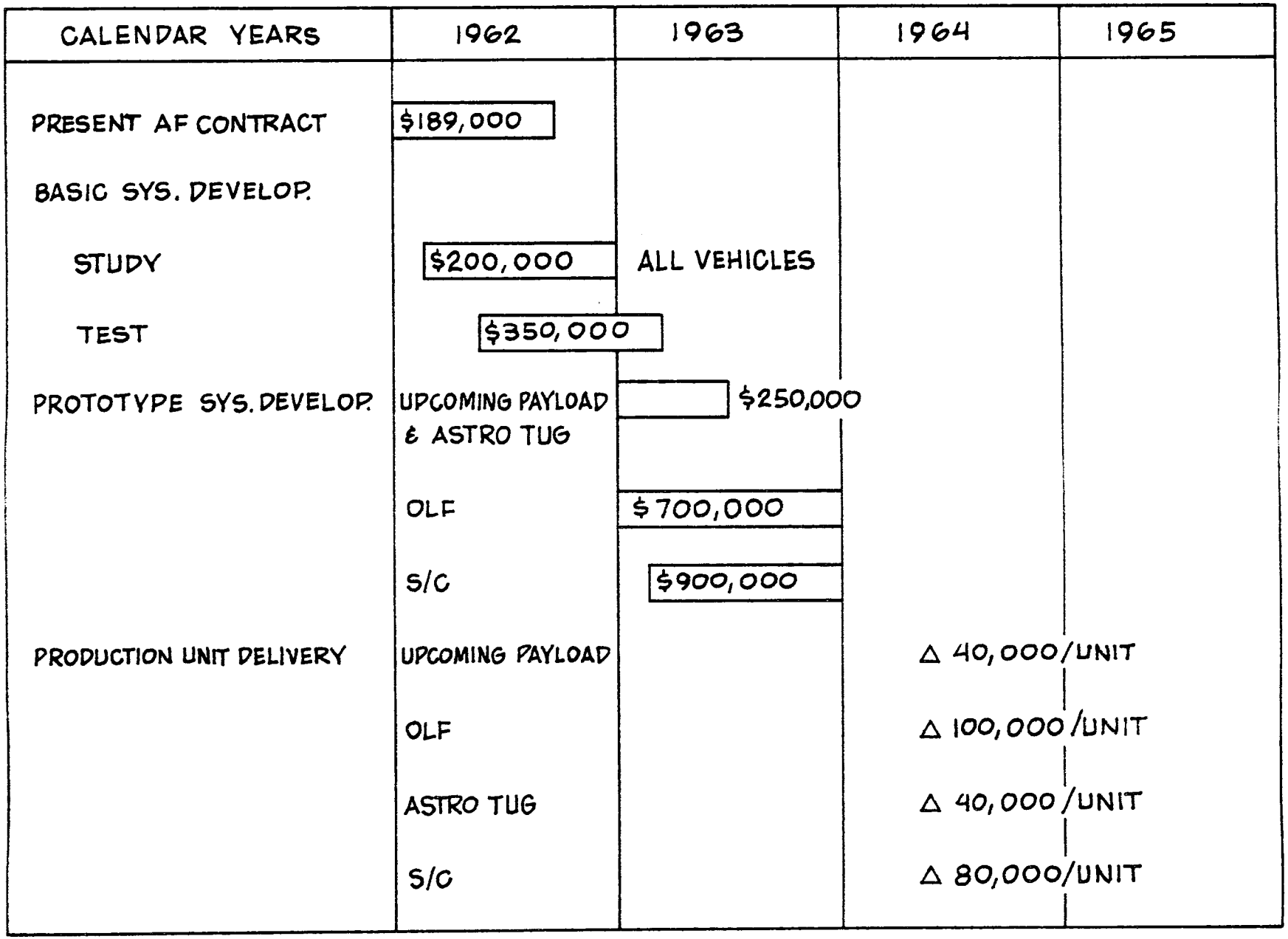
G. Program support

The R & D work on a basic power system operating fuel rich (shown as Phase I-B of the development program) is considered a necessary transitional step between the present engine program and the complete advanced power system operating stoichiometrically. The development program schedule shows that the advanced power system could become operational by 1965. If an operational system is required before that time, efforts could be concentrated on the basic fuel-rich power system which could be made operational at least one year earlier. With an early contractual authorization and sufficient support, Vickers believes that the basic Hydrox engine power system could be made operational by the end of 1963.

# PROGRAM SCHEDULE AND COST, STOICHIMETRIC SYSTEM



# PROGRAM SCHEDULE AND COST, FUEL RICH SYSTEM



## 6.0 - PROBLEM AREAS

The problem areas associated with a cryogenic hydrogen-oxygen internal combustion engine are listed below. These criteria will require the most development and will prove to be the limiting factors with respect to engine reliability, cost, complexity, and economy of operation.

1. Lubrication
2. Valve and injector mechanism design
3. Combustion

A wealth of background information, much of which is directly applicable, can be obtained from the literature on air-breathing internal combustion engines. A high degree of mechanical sophistication is permissible in a well developed system if enough redundancy is used to ensure reliability.

Some solutions to the problems mentioned above are as follows:

Lubrication - High temperature lubricants such as polyphenyl ethers are being investigated for operation in the two-stroke-cycle internal combustion hydrogen-oxygen engine currently under development at Vickers. The results of this investigation will be directly applicable. Open-cycle and closed-cycle lubrication will both be considered, and a choice will depend on lubricant consumption and relative heat transfer.

Injector and Valve Design - High inlet temperature and pressure, adverse valve and/or injector operating environments, and stringent

operating time and tolerance requirements create a difficult material and mechanical design problem for a propellant inlet mechanism. This problem is under investigation at Vickers in the program mentioned above. A high level of design sophistication in these areas is permissible, and much design experience is available.

Combustion Problems - Difficulty in the combustion of hydrogen in air has been experienced and the use of antidetonants is not desirable. However, variations in inlet pressures and mixture ratios and the possibility of premixing all or part of the combustible mixture leaves a large degree of latitude in the problem approach. This again points to the desirability of a well developed engine which can be extensively tested on land prior to incorporation into the system.

Temperature - The primary difference between the simple and the advanced engines lies in the operating and propellant inlet temperatures. (Both operate at a Carnot cycle efficiency of 0.74). An extensive investigation into the use of exotic materials, construction methods, and lubricants will be necessary to ensure reliability at the proposed stoichiometric combustion temperatures. Such an investigation would be more fruitful for a reciprocating positive displacement engine than for its turbine counterpart, for two reasons:

1. Peak cycle temperatures in a reciprocating engine can be relatively high with moderate average temperatures.
2. Operating speed and stress levels are much lower in reciprocating engines than in turbines, relaxing material strength requirements.

## 7.0 - CONCLUSION

Vickers has invested close to half a million company dollars over the past few years in the study and development of cryogenic auxiliary power systems. Optimization studies indicated that significant weight advantages were offered by a hydrogen-oxygen system using a positive-displacement engine. This advantage is derived from the superior characteristics of hydrogen as a fuel and the capability of the positive-displacement engine to operate efficiently at varying loads.

This same engine system is adaptable to closed-cycle operation utilizing solar or nuclear power for the long-range space vehicles. A study of the various types of expanders for power systems of this type indicates that the positive-displacement, variable-admission-ratio engine is more practical than other units contemplated for these missions.

An extension of the hydrogen-oxygen engine into a complete power and environmental system for space vehicles is foreseen in near-future development. The cryogenic hydrogen and oxygen perform several functions. Not only are they employed to drive the hydraulic pump and alternator, but also for crew compartment cooling, equipment cooling, heating, and attitude control.

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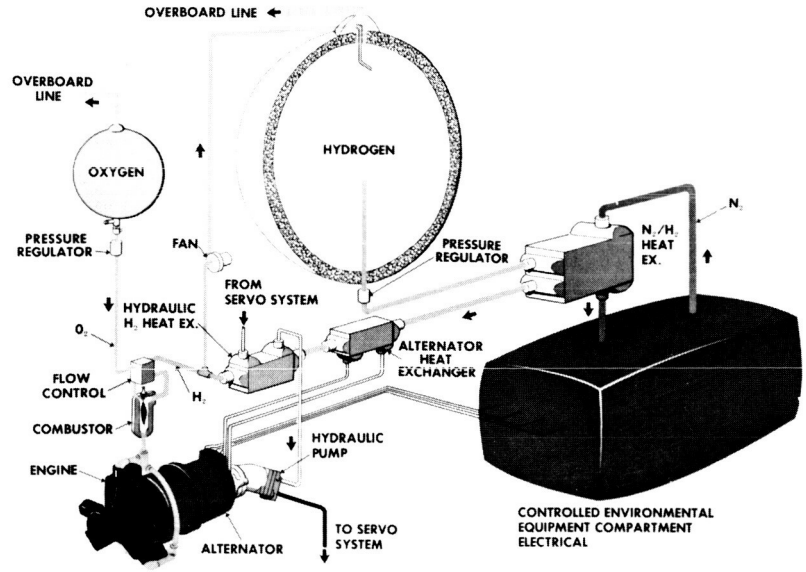



FIG. 3-1 - SYSTEM SCHEMATIC

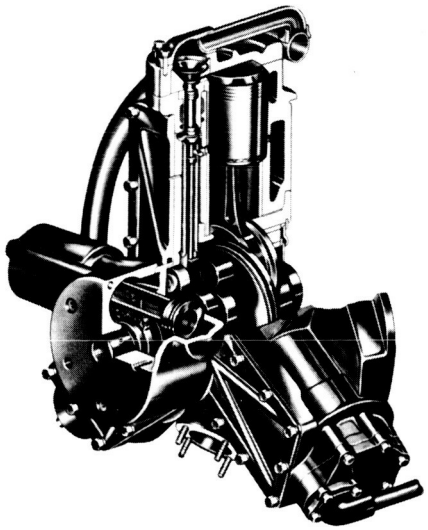


FIG. 3-2
POSITIVE-DISPLACEMENT ENGINE

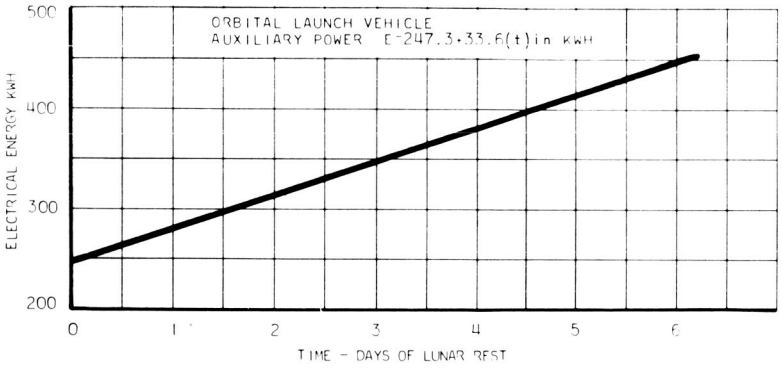



FIG. 3-3 - ELECTRICAL APS ENERGY REQUIRED IN TERMS OF LUNAR REST

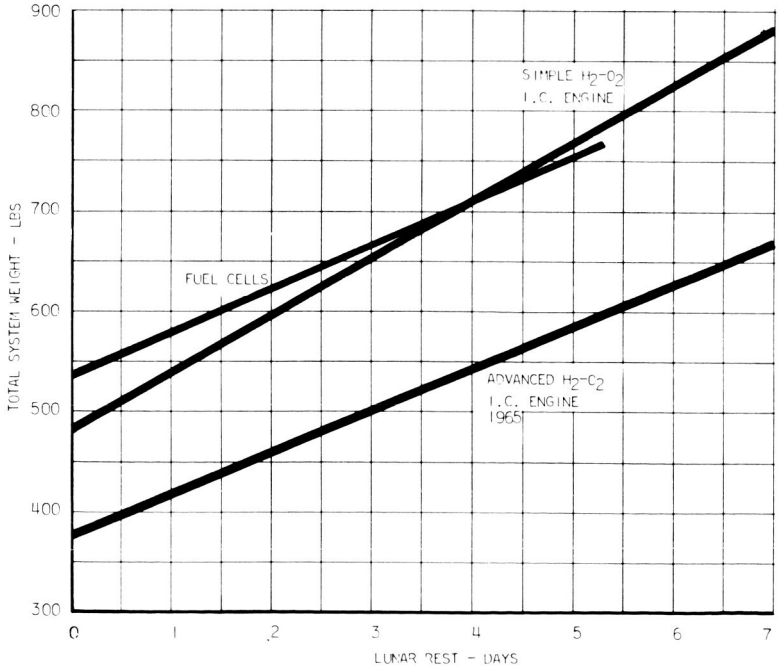



FIG. 3-4 - WEIGHT VS. DAYS AT REST

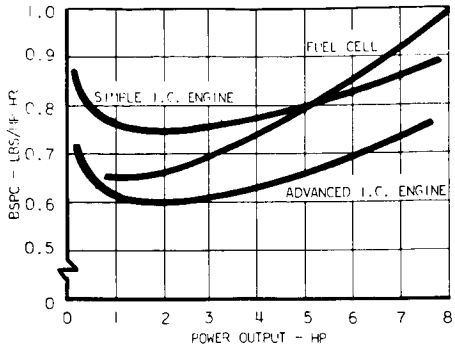



FIG. 3-5 - COMPARISON OF BSPC TO POWER OUTPUT

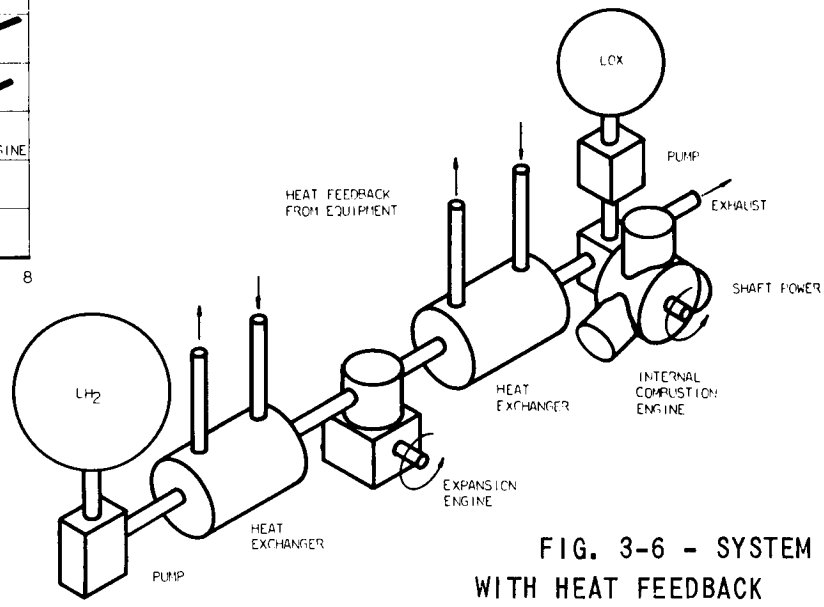



FIG. 3-6 - SYSTEM WITH HEAT FEEDBACK

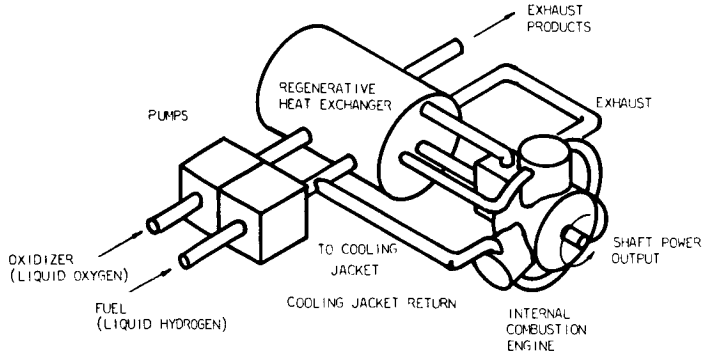



FIG. 3-7 - INTERNAL COMBUSTION REGENERATIVE POWER SYSTEM

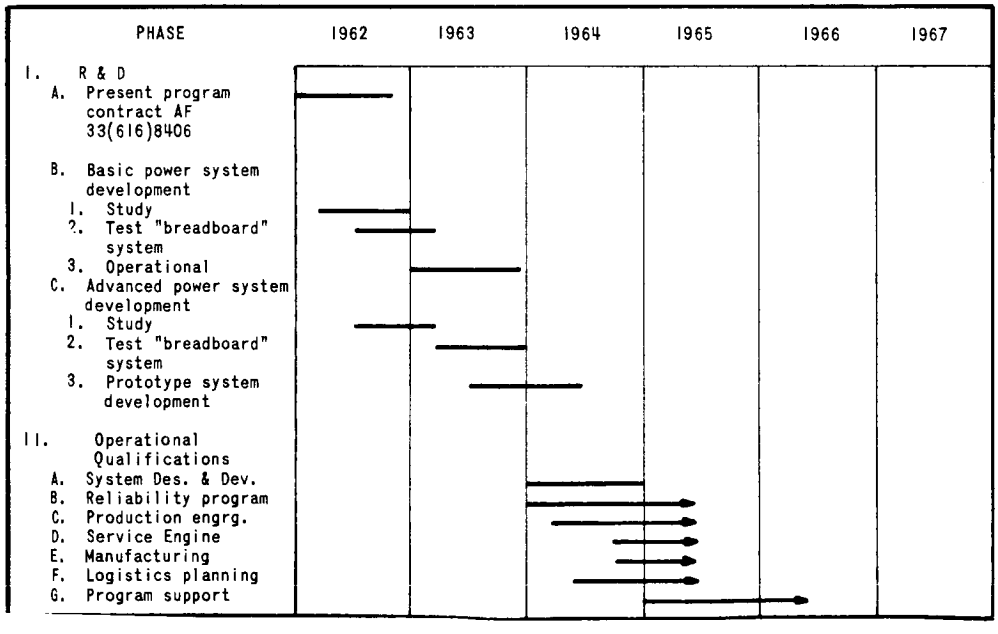



FIG. 3-8 - DEVELOPMENT PROGRAM SCHEDULE

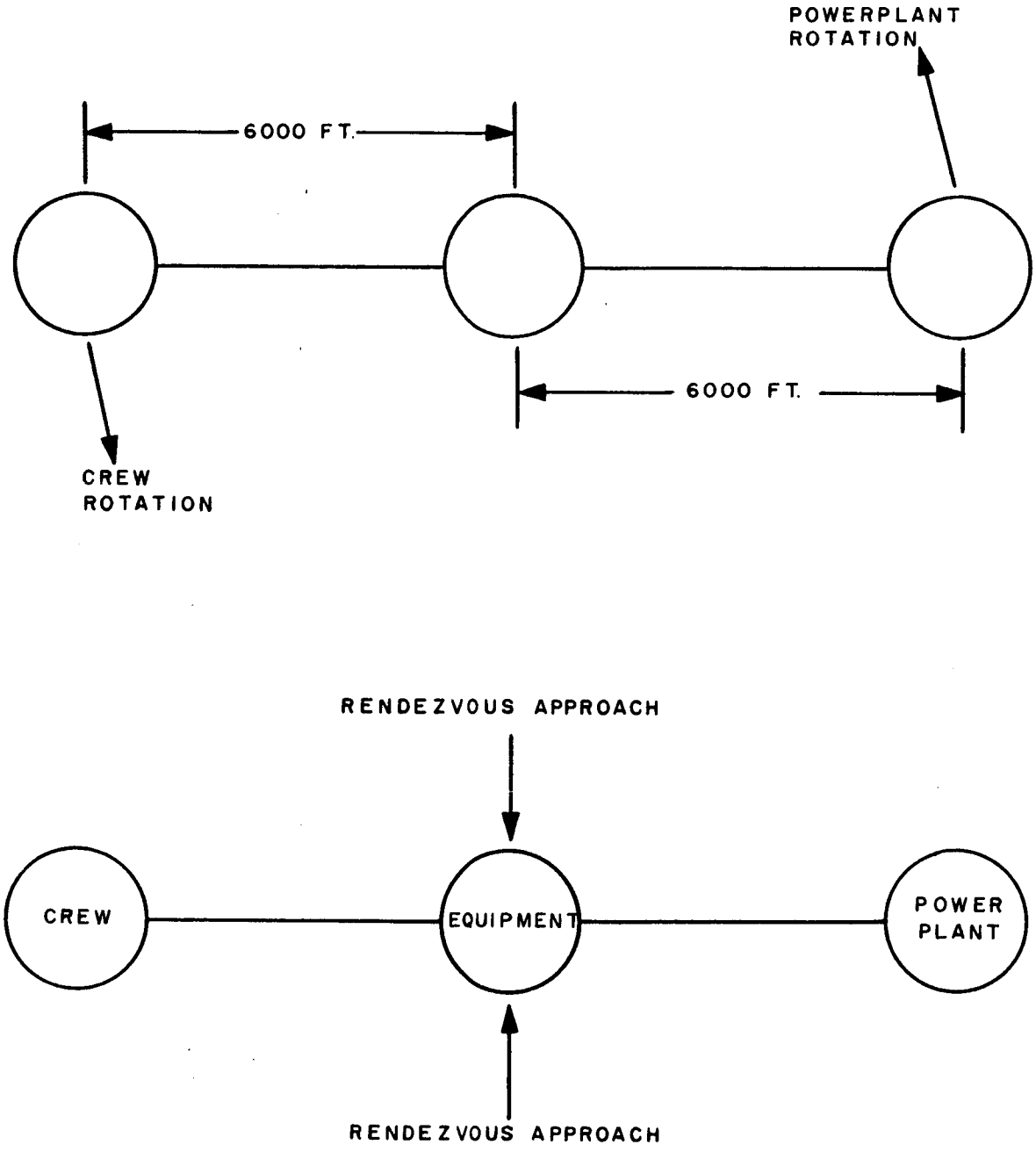



FIG. 3-8A. ORBITAL LAUNCH FACILITY WITH SAFE ARTIFICIAL GRAVITY

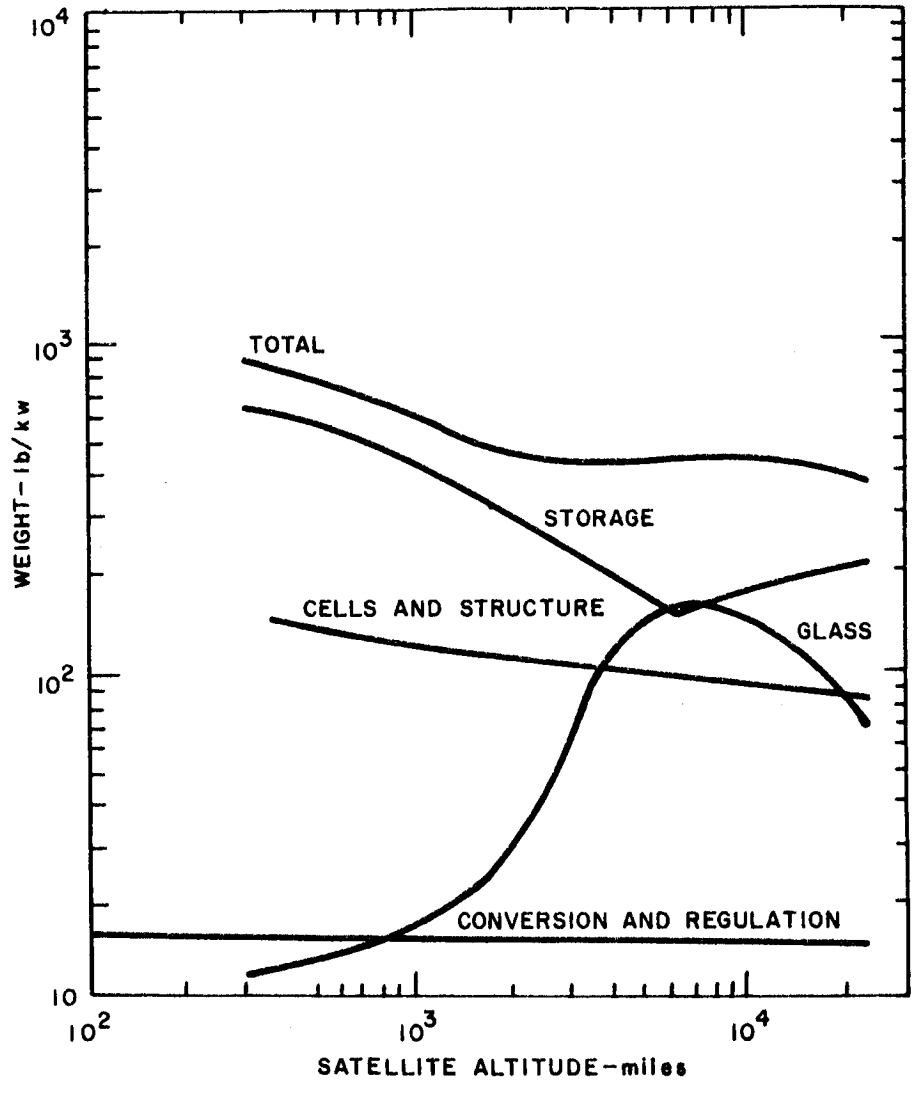



FIG. 3-8B. PHOTOVOLTAIC SYSTEM WEIGHT - 1962

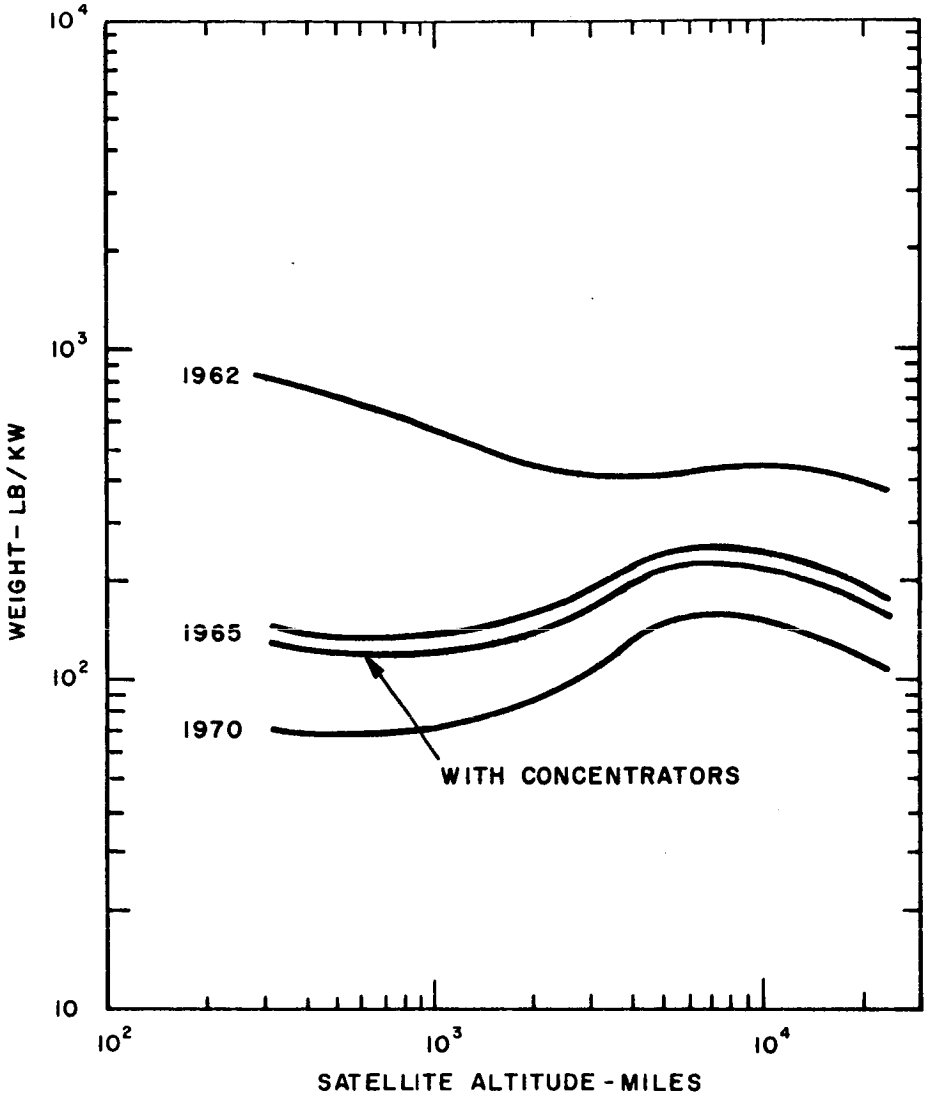



FIG. 3-8C. PHOTOVOLTAIC SYSTEM WEIGHT

## APPENDIX A - SUPPLEMENTARY SYSTEMS

Hot Gas Motor APS - Although hydraulic pumps driven by electric motors or flywheels are satisfactory for many low-power, short-duration applications, Vickers has placed accelerating emphasis on the hot gas power concept because of its many attractive advantages in space.

Concept - The Vickers hot gas motor APS consists of a pressure-regulated solid-propellant gas generator powering a modified Vickers positive-displacement motor that drives a standard Vickers pump. Shown schematically in figure 3-9, this system combines the advantages of compact propellant energy with the inherently high efficiencies of positive-displacement machines driven at moderate rotating speeds. A preliminary study of power versus specific weight is presented in figure 3-10.

Design - A photograph of the hot gas APU (integrated motorpump package) is presented in figure 3-11. The hot gas pressure regulator that was developed for the system is shown in figure 3-12. The pump is a standard unit with approximately one-half the displacement of the motor so that 1500 PSI gas pressure can be converted into 3000 PSI hydraulic pressure.

Several modifications were necessary to convert the basic Vickers hydraulic motor for efficient and reliable operation with high temperature gas. New materials with high wear resistance and nearly identical thermal expansion characteristics were selected for the cylinder block and pistons. The shaft seal was omitted, and a special valve plate was provided to withstand friction and temperature conditions.

The gas generator is of simple cylindrical construction, incorporating

an igniter squib and rupture diaphragm. The generator is designed to use solid propellants, including the single-base ammonium nitrate and the double-base nitroglycerin nitrocellulose. Many tests have proven the suitability of using either type of propellant charge. Motor inlet filtration has been found unnecessary using these propellants since the only contaminant produced by combustion is a soft carbonaceous substance that does not impair motor operation.

In addition to the motorpump APU concept, Vickers has developed motors that operate exclusively on hot gas to power mechanical devices. One of these motors is discussed later under "Applications".

Operation - An electrical signal to the gas generator igniter initiates propellant combustion. The grain combustion rate is constant at the constant pressure maintained by the pressure regulating valve. The high pressure gas (e.g. 1500 psi) preloads a hydraulic accumulator and the motor inlet. The accumulator thereby maintains gas system pressure in the pump motor case. The motor outlet is open to atmosphere, and case pressure acting behind the motor pistons causes them to reciprocate by virtue of the differential pressure across the motor inlet and outlet ports (i.e., the opposing pressures on the pistons on the inlet side are balanced, whereas a 1500 PSI differential exists on the outlet pistons). The motor drives the pump through a common shaft, producing hydraulic power.

The unique feature of driving motor pistons from the back side prevents gas "blow-by" past the pistons since opposing pressures are equal. Any oil leakage through the motor outlet is negligble compared with gas

leakage under the same conditions, and volumetric efficiency is optimized. This feature also provides a sink to dissipate heat from the motor components.

Applications - Vickers and the United States Air Force invested over a half-million dollars in a development program with the conviction that the hot gas motor auxiliary power system (APS) concept represented a significant advance in the state of the non-propulsive power art. A contract was awarded to Vickers by the Thiokol Chemical Corporation, working on the control and propulsive systems for the Minuteman ICBM. An accelerated research and development program was initiated to produce a compactly integrated APS within a seven-month period.

Twelve breadboard units were built, and ninety-one tests were conducted in 30 days to establish motorpump reliability. Twelve prototype APS systems were then fabricated to substantiate system reliability through a series of 123 operational tests. The flight hardware system is shown in figure 3-13.

Aside from testing of the self-sufficient APS, a program was initiated to establish the feasibility of operating a motorpump auxiliary power unit (APU) with gas bled directly from the Thiokol rocket engine. A successful bleed-gas test was conducted utilizing a Thiokol TU-132 engine to power the hardware. A peak of 2.15 hydraulic horsepower was developed after the hydraulic flow reached steady-state of 0.54 seconds and remained at 5.2 GPM for the duration of the test. Clogging of the gas pickup terminated the test after 16 seconds of operation.

The experiment was so successful that Thiokol sponsored a second feasibility test performed several months later with the test rig shown in figure 3-14. Results of the second test definitely established the feasibility of operating the Vickers APU at high temperatures with highly contaminated gas.

The motorpump ran for the full 62-second burning duration of the Thiokol main engine propellant charge. Motor inlet temperatures were recorded up to 3600°F with gas solid-contamination estimated at 10 to 15 percent by weight. A breakdown inspection of the motorpump revealed all parts to be in excellent condition (including the motor pistons which moved freely in the cylinder bores). A peak output of 2.35 horsepower was developed during this test with a steady-state hydraulic flow of 5.2 GPM.

Additional laboratory testing was conducted to simulate maximum temperature bleed operation using a near-stoichiometric bipropellant mixture of hydrogen and oxygen. The APU operated for 39 seconds at propellant flame temperatures estimated between 5000° and 5200°F, until a stainless steel line directing gas from the combustor to the motor ruptured. The motor components were found in excellent condition as shown in figure 3-15.

The hot gas motorpump shown in figure 3-16 is currently being used on the Skybolt missile. The 100-horsepower hot gas motor presented in figure 3-17 was produced for General Electric. Delivery of prototype hardware required only ten weeks (sixty days) from receipt of the specifications. Basic design, testing and modifications, final prototype assembly, and delivery were accomplished within this period.

This unit has undergone an acceleration rate of 50,000rpm/sec. in the G.E. application at essentially constant torque over its entire speed range. The motor is designed to operate on either a double-base solid propellant or on a liquid monopropellant, although all runs to date have been powered by the double-base solid propellant.

Battery-Powered APS - Although the potential applications of battery powered auxiliary power systems are limited in the OLO concept, they may be suitable for any short duration boosters associated with the program. A brief discussion of this approach to providing auxiliary (or primary) hydraulic power is included.

Concept - A schematic diagram of an integrated battery-powered hydraulic system is presented in figure 3-18. The essential components of the system are silver-zinc batteries, a d-c motor driving a hydraulic pump, a fluid reservoir, and valves. A preliminary study of power versus specific weight is presented in figure 3-19.

Design - Many APS designs based on this concept employ the "building block" principle whereby "off-the-shelf" electric motors are mounted to "off-the-shelf" hydraulic pumps. Where space and weight parameters require a more sophisticated approach, complete integration of the APS components can be achieved as shown in the schematic diagram.

A photograph of an integrated battery APS used to provide hydraulic power for the Minuteman missile is shown in figure 3-20.

Vickers has been producing airborne electrical motorpump packages for over 15 years.

Applications - Other recent applications include d-c motorpumps used to power hydraulic systems for the Pershing, Titan, Bomarc, and Polaris missiles. The Polaris motorpump, shown in figure 3-21 is a completely integrated unit with a special uncured latex coating that absorbs ambient heat (in curing), thereby protecting the unit from its high temperature surroundings.

Flywheel APU - Concept - Flywheel-powered APU's are attractive for many specialized applications in an orbital launch operation. The system consists essentially of a flywheel, hydraulic pump/motor, reservoir, and valves. These elements are related as shown in figure 3-22.

By using the pump/motor concept, the flywheel absorbs energy whenever system pressure rises above the nominal value and delivers exact quantities of power when the pressure falls below that value.

Design - The highest specific energy storage capability for a given maximum flywheel stress is theoretically achieved with a uniformly stressed rimless disc. However, this design requires top speeds that cannot normally be attained within practical size and speed limits. Furthermore, windage losses are highly sensitive to speed with a uniform-stress disc, dissipating a large proportion of the total stored energy in most applications. These considerations have led to the heavy rim flywheel for the unit shown in figure 3-23 that was designed for the Atlas missile.

The flywheel for this design is shrouded to accommodate air evacuation, minimizing windage losses. The pump/motor employed is essentially a standard fixed-angle variable delivery unit equipped with a special valve block to enable two-way operation (pumping and motoring).

A generalized weight study comparing specific system weight (not including flywheel weight) with rated output power is presented in figure 3-24. The flywheel weight depends on operating duration. Based on a tip speed of 1500 fps and allowing for windage losses, the duration factor for a flywheel is 150 lb/hp-hr. For example, a ten-horsepower system of five-minute duration will weigh 135 pounds (125-lb flywheel). A similar system for five seconds duration will weigh only 12.1 pounds (2.1-lb flywheel).

Operation - During run-up, hydraulic power is used to accelerate the flywheel to rated speed. This is accomplished by driving the flywheel-mounted hydraulic pump as a motor, using power either from ground supply or from vehicle supply (if the flywheel system is to be used later in the mission).

The flywheel-mounted hydraulic pump comprises a unit that has the characteristics of a highly efficient accumulator. A constant-pressure hydraulic output is achieved if the pump/motor unit is pressure-compensated to provide variable delivery, thereby compensating for flywheel energy degradation.

Application - The flywheel APU is particularly attractive in staged vehicles. For example, a flywheel system can be installed in the second (or third) stage of a vehicle, and the first-stage primary hydraulic system can "charge" the flywheel APU during any period before first-stage jettison when hydraulic demands are low. The second stage is thereby endowed with an independent hydraulic power supply without paying space and weight

penalties imposed by fueled prime movers or batteries.

As mentioned previously, the flywheel APU can also be used as an accumulator in a constant pressure system where the average hydraulic power demand is low compared with intermittent peak requirements. The flywheel is "charged" during low power demand periods and is available to supplement the main hydraulic power supply when needed. The primary advantage over other accumulator devices is that compensation for energy degradation can be achieved by using a variable displacement pump/motor unit.

Availability - Since the flywheel APU is composed of relatively standard equipment, the only development necessary to achieve a production unit concerns selection of a flywheel size and configuration compatible with the mission parameters.

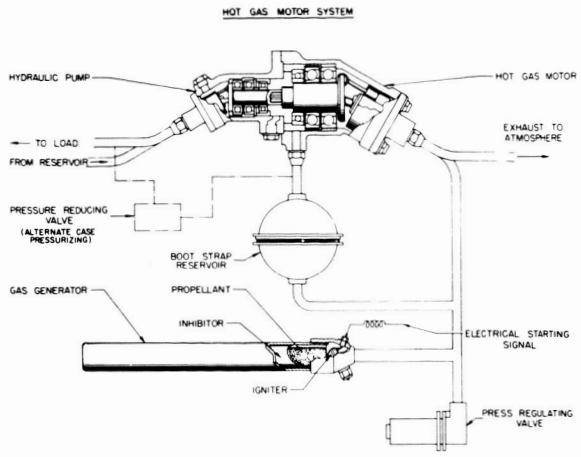



FIG. 3-9 - HOT GAS MOTOR SYSTEM

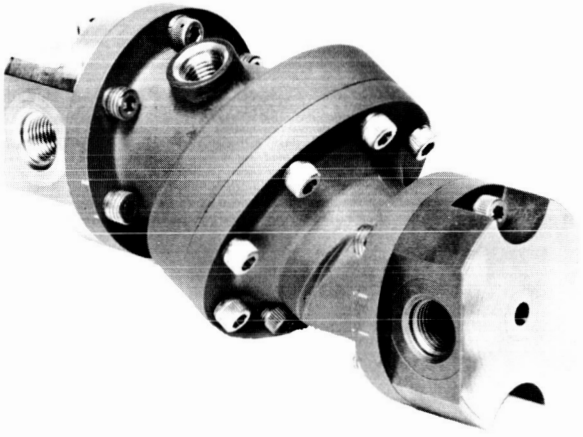


FIG. 3-11 - HOT GAS MOTORPUMP

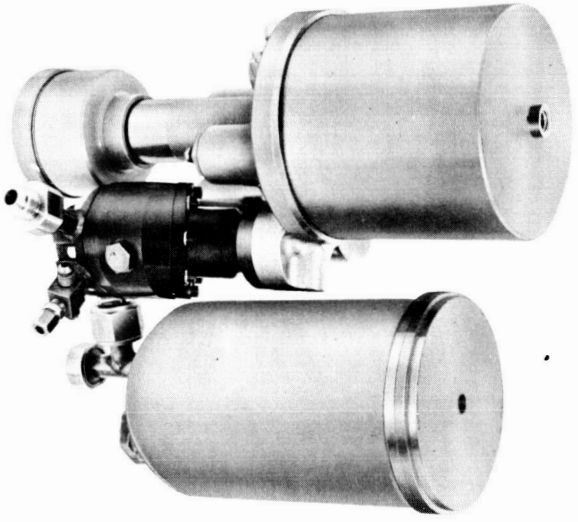
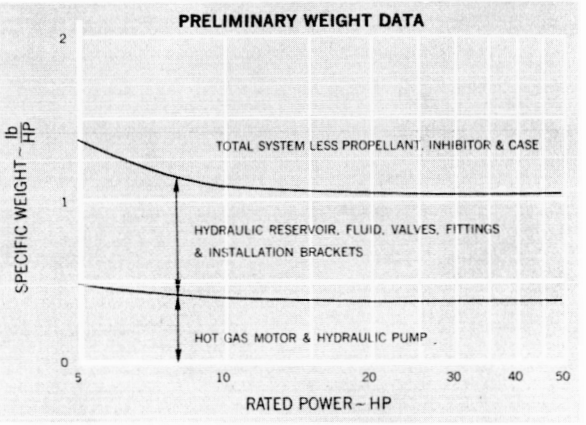


FIG. 3-13 - INTEGRATED HOT GAS APS




The duration factor for a short duration hot gas motor system utilizing a solid propellant is 27 lbs/hp-hr, which includes propellant, inhibitor and case weight. Therefore, as an example, a 10 hp system of 5 minutes duration will weigh 33.5 lbs. A 10 hp system of 5 seconds duration will weigh 11.4 lbs. For durations above 5 minutes, a partial admission motor with a duration factor of 18 lbs/hp-hr. would be used.

FIG. 3-10

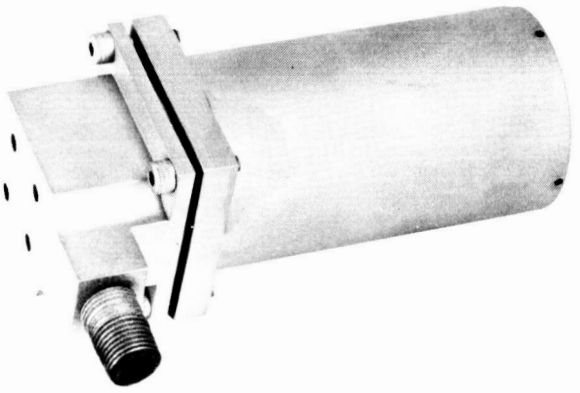


FIG. 3-12 - HOT GAS PRESSURE REGULATOR

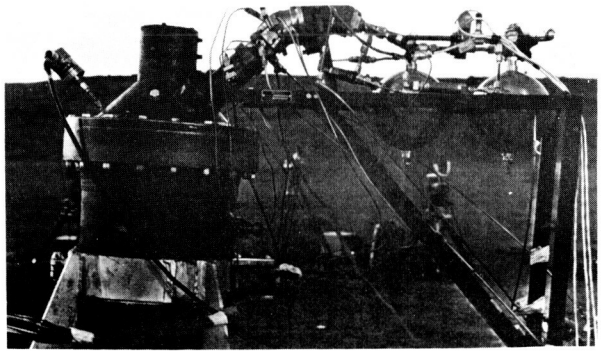


FIG. 3-14 - HOT GAS BLEED TEST

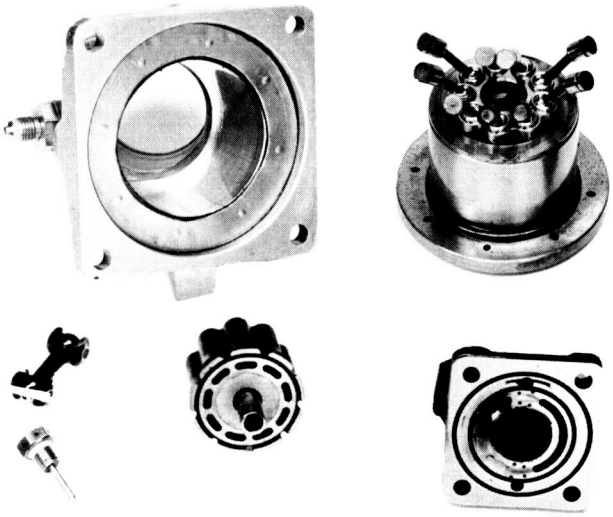


FIG. 3-15 - HOT GAS MOTOR AFTER OPERATION AT 5200°F

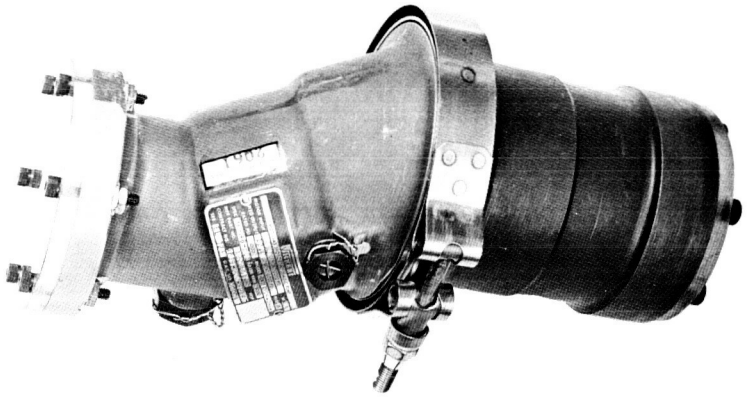


FIG. 3-16 - HOT GAS MOTORPUMP FOR SKYBOLT

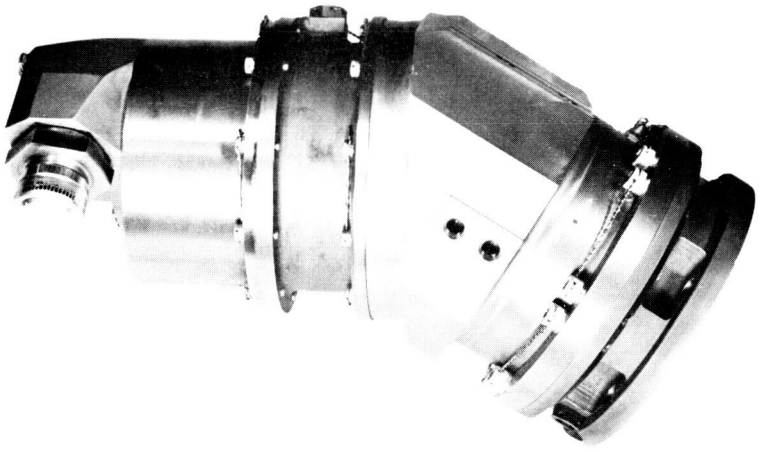


FIG. 3-17 - 100 HP HOT GAS MOTOR

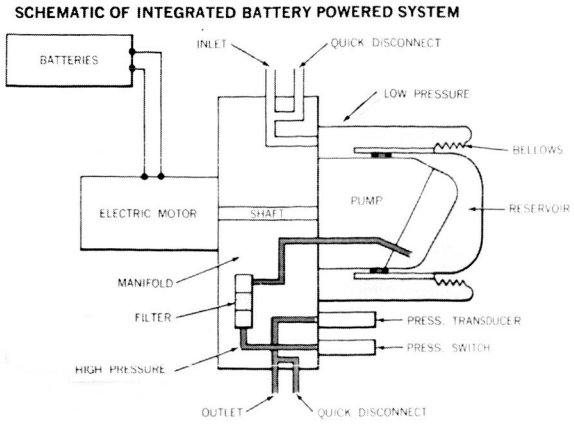
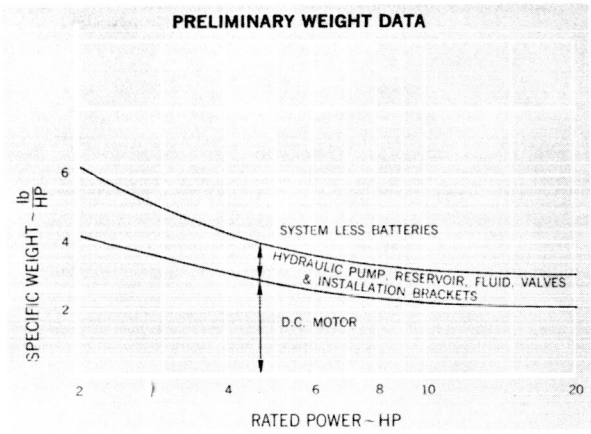



FIG. 3-18




The weight factor for a silver-zinc battery includes 1.0 lb/hp fixed weight and a duration factor of 15 lb/hp-hr. Therefore, a 1 lb/hp must be added to the d-c motor and hydraulic components specific weight shown on the curve at the left to obtain fixed system weight. As an example, a 10 hp system of 5 minutes duration will weigh 54.5 lbs. A 10 hp system of 5 seconds duration will weigh 42.2 lbs.

FIG. 3-19

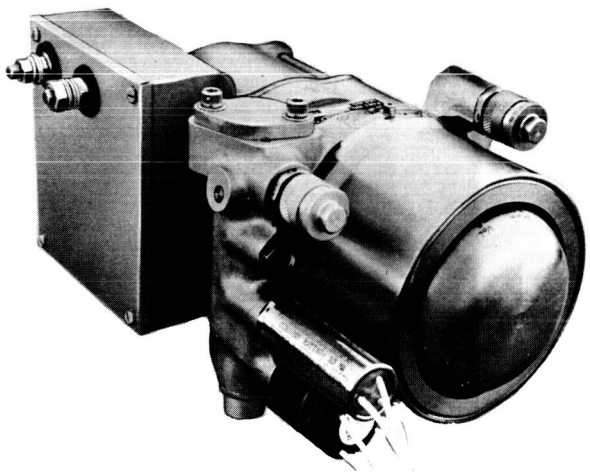


FIG. 3-20 - INTEGRATED BATTERY APS FOR MINUTEMAN

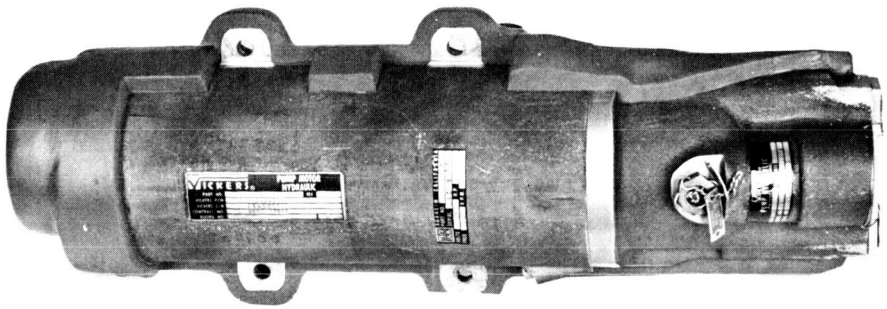


FIG. 3-21 - ELECTRIC MOTORPUMP FOR POLARIS

**SYSTEM SCHEMATIC**

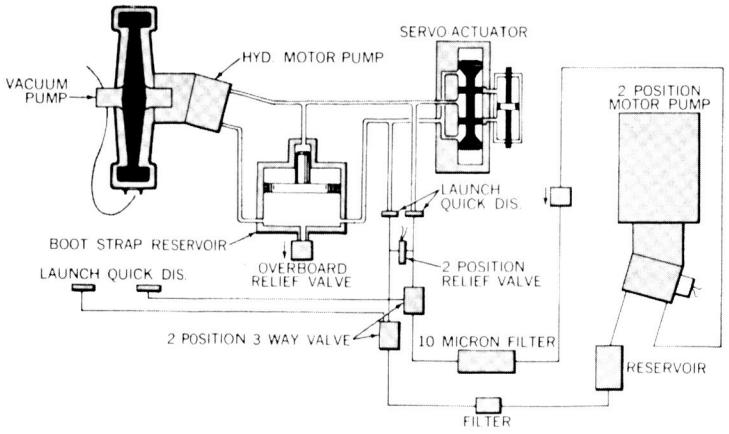



FIG. 3-22

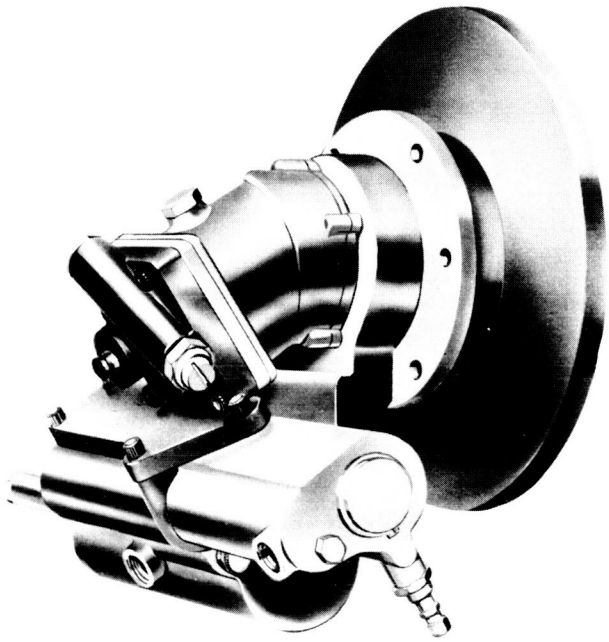
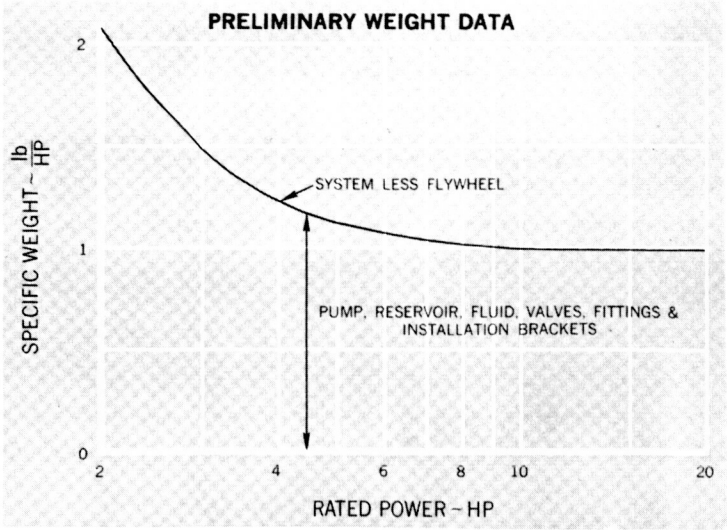


FIG. 3-23 - FLYWHEEL APU FOR ATLAS

FIG. 3-24



## APPENDIX B - DEFINITIONS

1. APS: Auxiliary or Secondary Power System; a device to generate hydraulic, mechanical or electrical energy for any components or subsystems other than main propulsion.

2. BSPC: Brake Specific Propellant Consumption (Lbs/HP-hour)

3. OLO: Orbital Launch Operations

4. OLV: Orbital Launch Vehicle; vehicle used to launch spacecraft from earth orbit.

5. OLF: Orbital Launch Facility; an earth-orbiting facility used for platform assembly and launch of spacecraft and OLV.

6. LOF: Lunar Orbiting Facility; a lunar facility that serves as the orbital terminal for lunar base supplies and personnel.

7. LDV: Lunar Descent Vehicle; vehicle used to shuttle cargo and personnel between LOF and Lunar Base.

8. MLM: Manned Lunar Landing and Return Mission

9. Apollo A: 3-man Apollo spacecraft control module with specific equipment required for suborbital and orbital flights.

   Apollo B: 3-man Apollo spacecraft, control, equipment and propulsion modules for circumlunar and lunar orbital flights.

   Apollo C: 3-man Apollo spacecraft, control, equipment and propulsion modules including all equipment required for the lunar landing mission.

   Apollo D: 6-man Apollo-type spacecraft.